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HELICOPTER PROPULSION SYSTEMS

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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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Helicopter Propulsion Systems

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No. 302

HELICOPTER PROPULSION SYSTEMS

Papers presented at the 57th Specialists' Meeting of the AGARD Propulsion and Energetics Panel, held at the Ecole Nationale Supérieure de l'Aéronautique et de l'Espace (ENSAE), Toulouse, France, on 11–14 May 1981

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PREFACE

A Specialists' Meeting on Helicopter Propulsion Systems was held in Toulouse, France, during the week of 11th May 1981. This meeting was the 57th conducted by the AGARD Propulsion and Energetics Panel. The objective was to bring together those individuals who have made significant contributions to the field and to promote dialogue on subjects related to component technology for turboshaft engines and transmissions, inlet protection systems, engine-airframe dynamic compatibility, and future requirements. This meeting was held in parallel with a Fluid Dynamics Panel Symposium on Aerodynamics of Powerplant Installations. The meeting site was the Ecole Nationale Supérieure de l'Aéronautique et de l'Espace, Complexe Aérospatial de Lespinet.

The Technical Evaluation Report (TER) for the Helicopter Propulsion Systems meeting was prepared by Mr Warner L.Stewart, Director of Technical Services, NASA-Lewis Research Center, Cleveland, Ohio. Mr Stewart's TER is a very complete summary of the meeting and will serve as a basis for future meetings on helicopter-related technologies.

The program committee appreciates the support given by the local coordinator and is pleased to have had the opportunity to hold the 57th meeting at such excellent facilities in Toulouse. In addition, the summary and TER provided by Mr Stewart are gratefully acknowledged. His work in support of the Propulsion and Energetics Panel has been valuable in high-lighting areas in need of increased attention.

JOHN ACURIO

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ACKNOWLEDGEMENT

The Propulsion and Energetics Panel wishes to express its thanks to the French National Delegates to AGARD for the invitation to hold its 57th Meeting in Toulouse, and for the personnel and facilities made available for this meeting.

CONTENTS

TECHNICAL EVALUATION REPORT

by

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1. INTRODUCTION

The helicopter emerged upon the aviation scene approximately 30 years behind fixed wing aircraft. Although it found some use during World War II, it was during the Korean War that large numbers of helicopters were built. Since then an increased number and types have emerged – to a point where, at the present time, it plays a significant role in both military and civil applications. Although in the past most helicopters were designed for military use, many designed specifically for civil or combined uses are emerging – responding to new needs for such an aircraft (off-shore oil, logging, etc.).

The significant improvements in helicopters have been paced by advancements in the propulsion system, including both the engine and power transmission. Indeed, the advent of the gas turbine engine revolutionized the helicopter much to the same extent as it did for the fixed wing aircraft. Its unique features of high power to weight ratio and reliability have resulted in it being used almost exclusively in present helicopters.

Although the modern helicopter and its associated propulsion system is highly sophisticated, many problems still persist. In addition, new requirements are demanding still further improvements in the technology of engines and transmissions. For these reasons – and the fact that the NATO community is depending upon the helicopter to increased degrees for both civil and military uses – it was very appropriate for the AGARD Propulsion and Energetic Panel to sponsor the subject Specialists' Meeting. The objectives were to provide a forum for discussion of current technical problems facing this sector, new requirements requiring advanced technology, as well as some of the new and emerging technologies of interest for this application.

2. SESSION CONTENT

The Helicopter Propulsion Systems Specialists' Meeting was divided into six sessions with a total of 25 papers planned. Three were withdrawn, with 22 papers actually presented. The Table of Contents includes those papers presented by session number. For purposes of this review, the order of grouping of these papers has been modified in order to better group them from a subject matter standpoint. These ten groupings will be briefly reviewed in relation to the meeting objectives.

General Interest – Two contributions were made of more general interest and nontechnical in content.

- A film was presented describing the pioneering efforts of Italy's Corradino d'Ascanio in piloting the first successfully controlled helicopter in 1930.
- Paper No.13, by D.Berthault (Fr), described helicopter development in France. He described this in terms of three generations or eras in helicopter evolution – piston, turboshaft and maturity. He pointed out that France now has 15 per cent of the world market through such aircraft as the Dauphin II, Super Puma, and Ecureuil. He ended on an optimistic note commenting on the increased emphasis in civil and multipurpose use as well as foreseeing increased collaborative efforts amongst the Europeans.

Surveys – Three papers were presented projecting a look into the future from the view of three countries.

- Paper No.1 by G.Holbrook (US), was the keynote address looking at the past, present, and future of helicopter propulsion. Issues for the future identified in this excellent paper included reliability, fuel consumption, life cycle cost, and emergency power rating. These issues became themes to be repeated many times throughout the meeting. The paper also presented some aggressive projections for the future and challenged the engineering community to make it happen.
- Paper No.4, by M.Paramour and M.Sapsard (UK), paralleled the keynote speech but with major emphasis on military issues. The paper identified environmental hazards, range and endurance, life cycle costs, and flexible use

of ratings as major concerns for the future. It also concluded that, for their applications, environmental resistance and low cost of ownership would be of greater importance than higher performance.

Paper No.25, by H.Bree and G.Backmann (GE), also related principally to military missions. It described mission needs and related them to propulsion requirements. It pointed out that two-engine aircraft would probably be required from the standpoint of flight training. It also commented on the impact of dwindling resources (fuel, and strategic materials) on the design of future engines. It also emphasized the overriding importance of life cycle cost (including both procurement and ownership aspects) for the German military missions.

Development History Two papers documented some of the experiences gained in recent development programs.

- Paper No.2, by J.Fresco (FR), reviewed procedures used to satisfy certification and airworthiness requirements for the Makila engine used in the Super Puma. Topics covered included simulation testing of control responses, endurance testing and salt corrosion qualification testing.
- Paper No.3, by C.Crawford and W.Crawford (US), discussed lessons learned during the T700 development program. The two part paper first covered the wide range of requirements placed on this engine to meet the US Army needs. The second part then described how successful the G.E. development program was in meeting these requirements. Its success was pointed out to be the result of a full test program during development, extensive service testing, and time to make fixes before committing the engine to production.

Engine Component Technology - Three excellent papers were presented directed at technology developments for small gas turbine engine components.

- Paper No.8, by J.Schrader and W.Schneider (US), dwelt principally on advancements in the aerodynamic components and discussed higher pressure ratio compressors with fewer number of stages, advanced reverse flow combustors offering reduced emissions, and cooled small axial turbine stages.
- Paper No.5, by J.Dominy and K.Hart (UK), concentrated on components for power transmission systems including bearing and engine gearing technology – and then internal air systems including turbine blade and nozzle vane cooling together with disc sealing and cooling.
- Paper No.24, by C.Walker, G.Weden and J.Zuk (US), complemented the above two papers by touching on all of the areas involved. Subjects covered included LDV systems for flow mapping, felt ceramic combustor liners, variable area radial turbines, and spiral groove lift-off seals.

The above papers emphasized challenges to the designer to confront the detrimental effect of reduced size on the performance and mechanical integrity of components for lower power turboshaft engines. They also highlighted the role and importance of advanced computational methods in the design and analysis of these advanced components.

Preliminary Development - Two complementary papers were presented representing some of the early development activities in Europe in relation to a medium size advanced turboshaft engine.

- Paper No.6, by J.Hourmouziadis and H.Kreiner (GE), presented MTU's part of the activity, looking at a 900 kw unit. The paper was in two parts. The first concerned itself with configuration selection working from a base to a considerably more simplified arrangement. Cycle pressure ratio of 12 was selected as the optimum when considering related losses. The second part then described some of the related component activities. The key to success was viewed as being a combination of good analytical tools, component test first, and then verification in engines.
- Paper No.7, by M.Giraud and H.Loustalet (FR), described their part of this program and, in a manner similar to the previous paper, included two parts. The first described the interactive nature of "architecture", component geometry, and cycle. The study started with the simplest of architecture but moved close to the German geometry when cycle considerations (SFC) were considered. The paper also described related component activities. The conclusions emphasized the importance of the integration of the three considerations described above, and also pointed out the requirement of collaborative development efforts.

Alternative Engine Cycles Although several papers touched on this subject, one was exclusively devoted to it.

- Paper No 9, by H.Grieb and W.Klussmann (GE), re-examined the potential of the recuperated engine with emphasis on military missions. It was pointed out that such a cycle is favourable when extensive operation at low "mean" power is utilized. Several layouts were presented and a reasonable potential was shown in terms of both weight (including fuel) and life-cycle costs.

Transmission Considerations - Three excellent papers spotlighted the importance of helicopter power transmission and covered a range of subjects from research to technology to development.

- Paper No.10, by B.Shotter (UK), was a fundamental paper dealing with lubrication breakdown between meshing gear teeth. Its intention was to give a feel for the physics of wear initiation including scuffing and micropitting – concentrating on the sources of damage. Illustrative photos were included together with sketches to show contact history in relation to film wedge buildup and breakdown.

– Paper No.11, by K.Rosen and H.Frint (US), covered two technology programs. The first considered high contact ratio gearing and included descriptions of both the advantages of this type of gearing as well as attending problems. The second part then highlighted progress made in exploring the potential of a fabricated stainless steel transmission housing. Comments on recommended future transmission RNT were also included.

- Paper No.12, by A.Garavaglia and G.Gattinoni (IT), discussed the design criteria for the A129 helicopter drive system. The paper showed some of the interesting and novel engineering features included in that system.

Inlet and Protection Systems Two very complementary papers were presented covering this subject which is of extreme importance to helicopter operations.

– Paper No.14, by A.Vuillet (FR), discussed the many factors affecting helicopter inlet design (ice, sand, gas ingestion, pressure distortion and loss, etc.). Comparisons between "static" and "pitot" inlets were covered at length including forward velocity effects. Aerodynamic considerations required included entrance position, surface area, and lip thickness.

- Paper No.17, by P.Brammer and D.Rabone (UK), described case histories in the area of environmental protection. Experiences with the Sea King were highlighted with the various design modifications to incorporate added protection against the elements described. A similar discussion of the Lynx was included. Proposed protection for the T700 powered EH101 was also included and indicated to be very conservative.

Both papers indicated that the subject of inlet protection is a very difficult problem indeed and must be considered as an important f tor in the beginning of development to insure success.

Propulsion/Airframe Compatibility - Three papers were presented covering three important aspects of this subject.

– Paper No.20, by C.Albrecht (US), emphasized compatibility attention required due to the sophistication of this type aircraft, the cost of correcting problems, as well as the myriad of interfaces involved. Several interesting examples were cited including both the problem and solution. The paper indicated that, to minimize such problems, the latest in technology must be applied together with a strong propulsion/airframe company relationship.

- Paper No.21, by D.Dini (IT), presented an interesting treatise on factors affecting engine response due to main rotor transients. Engine effects considered included gyroscopic, acceleration, and distortion. Torque input variations were described from such sources as lift variations due to forward velocity, types of rotor systems to be considered, as well as flight maneouvres.

– Paper No.22, by G.Genoux and H.Maize (FR), dwelt on two very significant aspects of helicopter design - vibration and noise. It included two parts. The first discussed rotor generated vibration and described methods to reduce this vibration at the source as well as techniques for reducing its transmission into the fuselage. The second part dealt with main gear noise considered the most significant source of cabin noise. Factors considered in reducing this noise included gearing redesign, special oils, reduction of resonances through the transfer chain, damper utilization, and sound absorption treatment.

Emergency Power Rating Although touched on in many papers, two papers were devoted exclusively to this extremely important and critical subject. Of course, of most significance is the application to twin-engine installations.

Paper No.19, by J.Dedieu, M.Russier and H.Dabbadie (FR), gave an approach introducing the concept of "superemergency rating" which translates into the utilization of higher turbine inlet temperature during emergency use. Engine application studies were presented to show its attractiveness. The relation of over-temperaturing to hot section creep was then described together with a recommended new approach to engine utilization. This would include a one-minute limit at this high temperature with subsequent engine removal for inspection and possible overhaul. It was hoped that new certification procedures employing this concept would be incorporated in the near future.

Paper No.23, by D.Lewis (UK), also addressed this issue head-on. The paper described the present rating system together with its many disadvantages. It also introduced the term "creep" as a measure of accumulated hot section damage. The paper proposed the introduction of an engine monitoring system that would allow tracking of this "creep". Certification would then include power versus creep information and would be used to gauge life consumed.

3. CONCLUSIONS AND RECOMMENDATIONS

The papers presented at the meeting, in the Technical Evaluator's opinion, were very successful in accomplishing the objectives outlined at the beginning. The major issues and concerns related to helicopter propulsion were amply spotlighted. They were clearly presented, treated from several viewpoints, and included extensive discussion. In general, the papers were not heavy in terms of detailed technical content but certainly were at a depth sufficient for purposes of the meeting.

It is recommended that, for future meetings, the Propulsion and Energetics Panel consider some of the more critical subjects surfaced during the course of the meeting and covered by many of the papers. Suggested topics include the following:

- *Small Engine Technology* addressing issues of reduced size on aerodynamics, and mechanical performance, as well as structural integrity. The subject is important for applications beyond that of helicopters.
- *Transmission Technology* covering such elements as performance, reliability, vibration and noise. Extremely important in relation to increased acceptance of the helicopter in the civil market.
- *Inlet and Environmental Protection* which has become so critical for many of the more severe missions encountered by this type vehicle.
- *Emergency Power Rating* was a subject brought up repeatedly at the meeting as a critical issue in the optimizing of engines for use in two-engine helicopters (the one-engine out issue). Present rating systems should be thoroughly explored together with an examination of all the potential alternatives.

Finally, the Technical Evaluator would like to express his appreciation for the opportunity to serve in this capacity in contributing to the 57th Specialists' Meeting of the Propulsion and Energetics Panel. It was indeed a pleasure.

Helicopter Propulsion Systems - Past, Present and Future

by

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Keynote Address

AGARD - PEP Specialists Meeting on
Helicopter Propulsion Systems

Toulouse, France

May 11-15, 1981

SUMMARY

Helicopter propulsion systems are reviewed, and it is noted that helicopter development is paced to a major extent by the power plant. Power available, reliability, fuel consumption, power-to-weight ratio, and life-cycle costs are key parameters.

The application of emerging technologies such as microelectronics, ceramics and other new materials and approaches, and the continuing refinement of the aerodynamics and dynamics of gas turbine power plants are discussed and noted to result in a significant benefit to the helicopter and its operator.

Important airframe-propulsion system interface requirements are given, and the need is discussed for new innovative certification procedures that provide for emergency operation with acceptable economics.

Finally, future propulsion system capabilities are projected and their dramatic benefit for the helicopter noted. Bringing this about is the challenge for propulsion system specialists.

INTRODUCTION

A key element in the progress of rotary-wing aircraft is its propulsion system, considered here to consist of the engine and the rotor transmission, together with their respective interfaces to the airframe.

The early helicopters (Figure 1 of 2) were powered by the Otto engine for the same reasons that that engine was first selected for fixed-wing aircraft. This engine could be made reasonably lightweight, had a low fuel consumption, was reliable and available in a range of power outputs that was adequate for the needs of the early aircraft. However, it became apparent that another type of power plant would be needed because the limiting power output per cylinder for reliable operation of the Otto engine was about 115 horsepower per cylinder, and that increasing the numbers of cylinders beyond eighteen led to unacceptable complexity and operational problems. With this type of engine, a transmission is required to reduce the engine output speed to that of the rotor with engine/rotor speed ratios of 9:1 to 12:1.

Figure 1 - Sikorsky Model R-4, First U.S. Production Helicopter

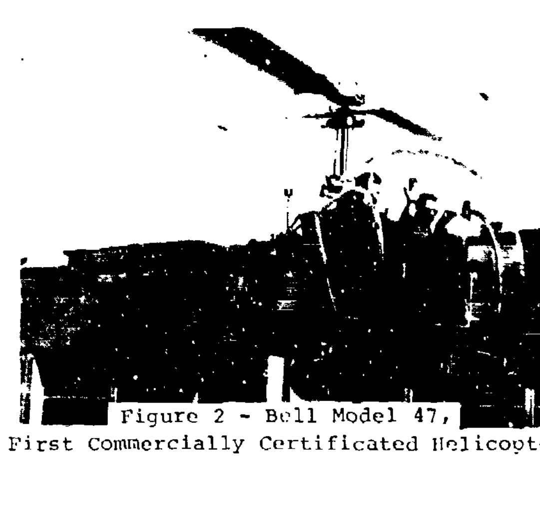
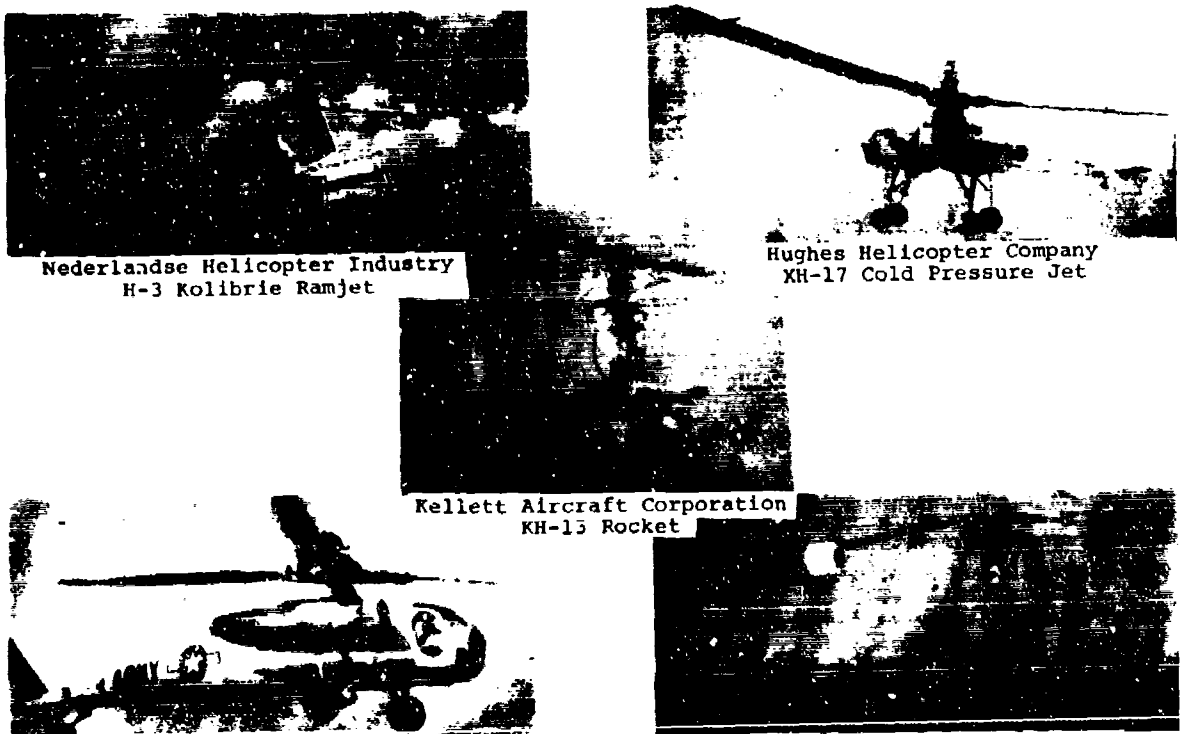


Figure 2 - Bell Model 47, First Commercially Certificated Helicopter

The transmission was considered to be a disadvantage principally because of weight and complexity, and a number of alternate types of power plants were investigated aimed at eliminating the rotor drive transmission, the attendant antitorque system and the inherent power limitation of the Otto engine. Experiments were conducted with rotor-tip-mounted low- and high-temperature air nozzles or pressure jets, ramjets, pulse jets, and even rockets. Most of these systems were built in a single rotor configuration and flown successfully. Figure 3 illustrates some of these early transmissionless machines.



Nederlandse Helicopter Industry
H-3 Kolibrie Ramjet

Hughes Helicopter Company
XH-17 Cold Pressure Jet

Kellett Aircraft Corporation
KH-15 Rocket

Hughes Aircraft Company
XV-9A Hot-Cycle Turbojet

American Helicopter Company
XH-26 Pulsejet

Figure 3 - Transmissionless Helicopters

It was found that while these power plant weights were relatively low, the complexity and structural problems associated with delivering compressed air and/or fuel and electricity up the mast and out the rotor blades, the high centrifugal force field in which blade-tip-mounted engines were required to operate, the noise they generated, and their very high fuel consumption made them impractical or, at least, not competitive for missions beyond about 50 nautical miles. Their fuel consumption ranged from about four times that of a reciprocating engine for a hot pressure jet to a factor of 50 times for a rocket.

Concurrently, the efficiency and reliability of the gas turbine engine were improving rapidly. The free power turbine, driving the rotors through modern transmissions, was ultimately adopted for most helicopter applications because of the inherent advantages of the turbine, its favorable torque-RPM characteristics, and its freedom of the need for a disconnect clutch. The power response characteristics of the two-shaft turbine were recognized to be inferior to that of the single-shaft turbine, and some helicopter manufacturers selected the single-shaft engine and accepted the penalties associated with a disconnect clutch. Power response is an even more important consideration for multiengine helicopters.

Current gas turbine engines have cruise fuel consumptions of about 1.2 to 1.5 times those of reciprocating engines of the same output. However, the reduced turbine engine weight offsets the slightly higher fuel consumption for all but the longest ranges. Additionally, the turbine operates with considerably less noise and vibration. With its use have come improvements in reliability both for the aircraft and the engine. Most current turbine engines require transmission reduction ratios in the 20 to 25:1 range for helicopter application.

As the turbine was introduced into the helicopter, design improvements in the second generation or evolved transmissions allowed the increased gearing associated with the higher ratio to be absorbed, essentially without notice. Allowable stresses of primary bevel gears were increased by 30 percent through the application of better materials and manufacturing methods. Transmission time between overhaul (TBO) was increased because of the experience developed with the reciprocating engine, continuing better designs, and due to the smoother operation associated with the turbine. The Bell Model 47 trans-

mission TBO was about 200 hours when it was first put into commercial service. When the Huey helicopter was at a comparable stage of development, its transmission TBO was 1000 hours.

Although significant progress has been made in rotor and airframe aerodynamics and structures, the high state of development of the helicopter would not have been possible if it were not for the introduction of the gas turbine with its high horsepower output per unit weight and volume, good efficiency, and its virtually unlimited power output potential.

Although there is still some interest in the pressure jet propulsion system for very large helicopters and a great deal of interest in the reciprocating engine for the very small helicopter, future progress in helicopters will be paced by the advancing technologies in gas turbine and transmission design and manufacture that will further increase their reliability, efficiency, and decrease specific weight, and volume, all at an acceptable cost.

It is the purpose of this specialists meeting to examine these emerging new technologies so that the potential improvements that they offer can be understood and used to improve the helicopter.

PRESENT STATE-OF-THE-ART

There are in service today a wide range of gas turbine-powered helicopters for both military and civilian applications. Governments most often sponsor the initial research and development efforts for a new engine and helicopter maunufacturers have been quick to adopt the new engines to improve their commercial helicopters, resulting in immediate profitable use by the operators. The helicopter manufacturers and the operators have jointly developed this new, vigorous industry, but it is paced by engine development.

In some cases, it is feasible to replace reciprocating engines with a gas turbine. The S58-T and the Soloy 47 are examples. The Soloy 47 (Figure 4) is an Allison Model 250 gas turbine repowered version of the 260-horsepower Lycoming piston-powered Bell 47, which went out of production in 1974. Benefits for this installation are improved performance and better direct operating cost achieved at an increased engine initial cost at least five times that of the reciprocating engine.

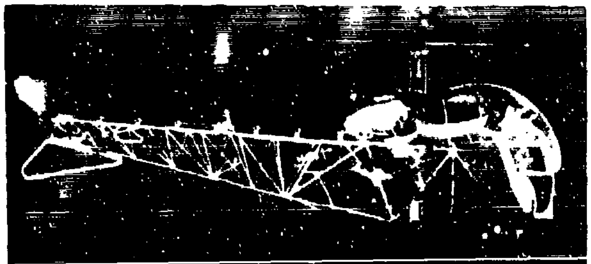


Figure 4 - Soloy Turbine Repowered Bell Model 47

There are now available a number of well-developed free turbine engines ranging from 420 up to 4400 horsepower that are specifically designed for use in helicopter propulsion systems. Figure 5 shows a map of the key parameters associated with available engines. The spread in parameters indicates the effect of size and technology level. Figures 6 through 12 illustrate some of the available engines and the helicopters in which they are installed. Other engines are under development to produce over 8000 horsepower on a 90°F day at 4000 feet altitude. There is still a lack of a flight-qualified turbine engine of under 400 horsepower rating. For this reason, a sizable number of helicopters of 3000 pounds gross weight or below, such as the new Robinson R22, continue to use the reciprocating engine.

The available engines may be used in single or multiengine installations for efficient helicopters with gross weights ranging from about 3000 to well over 100,000 pounds. Thus, except for the very small machines, helicopter development in almost any size class is <u>not</u> limited by lack of available turbine power plants. With most new commercial developments, however, it would be most unusual to find an available high technology power plant that would provide exactly the desired performance for the precise size helicopter that is sought. There is a different situation with the military where very careful attention is paid to providing the proper size engine, and where most often engines are developed for planned helicopters.

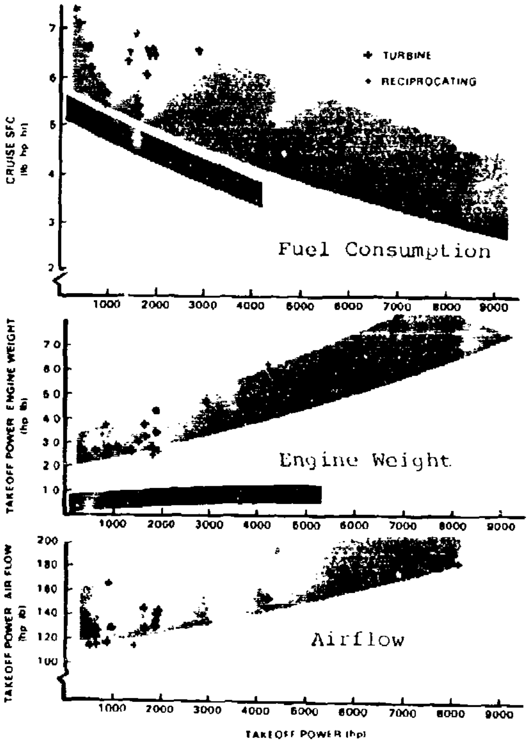


Figure 5 - Map of Key Engine Parameters

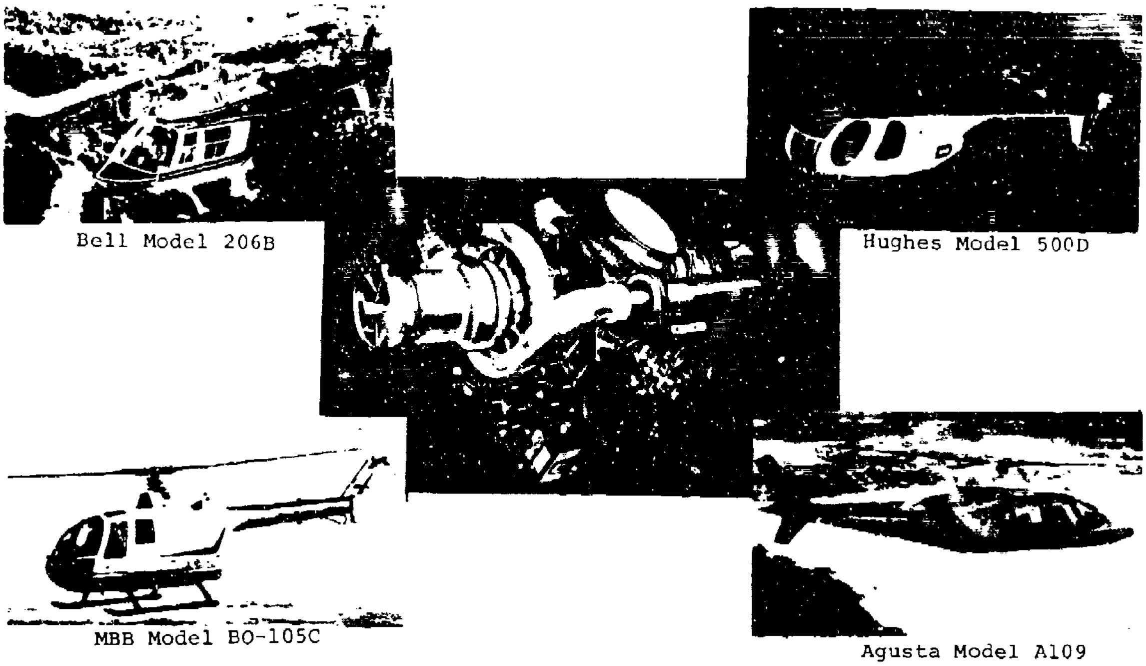
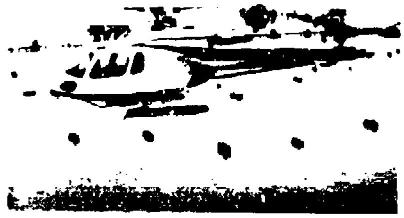


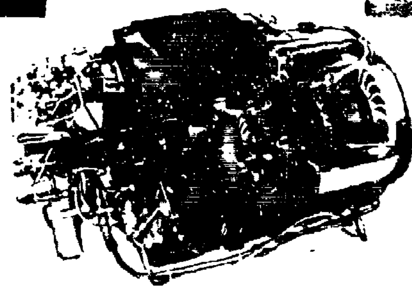
Figure 6 - Detroit Diesel Allison C20B Turbine Engine and Helicopters it Powers

Figure 7 - Detroit Diesel Allison C30 Turbine Engine and Helicopters it Powers

Bell Model 222



Aerospatiale Model AS350

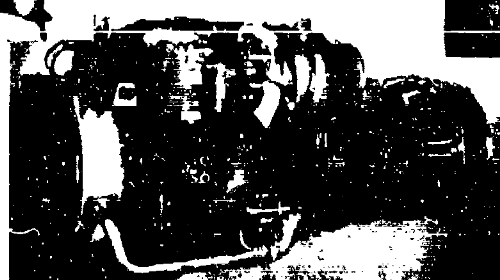


MBB/Kawasaki BK117

Figure 8 - Lycoming LTS 101 Turbine Engine and Helicopters it Powers

Hughes Model AH-64

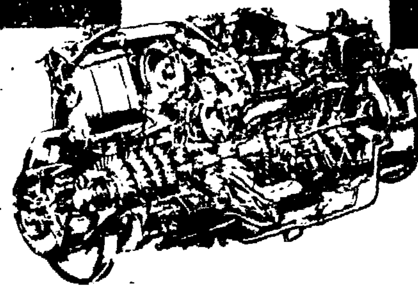
Bell Model 214ST

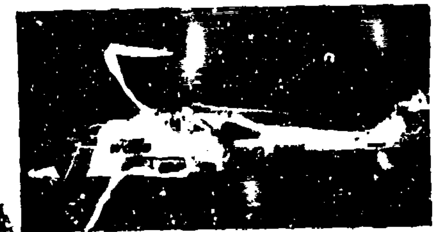


Sikorsky Model UH-60

Figure 9 - General Electric T-700 Turbine Engine and Helicopters it Powers

Agusta Model A129





Westland Model WG30

Figure 10 - Rolls Royce Gem Turbine Engine and Helicopters it Powers

Aerospatiale Model 332

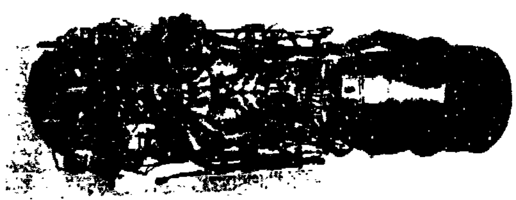
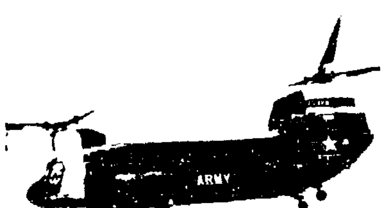


Figure 11 - Turbomeca Makila Turbine Engine and Helicopter it Powers



Boeing Vertol Model CH-47

Bell Model 214A/B

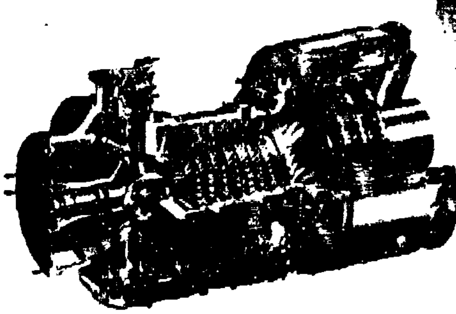


Figure 12 - Lycoming T55 Turbine Engine and Helicopters it Powers

Power, weight and fuel consumption data from current engines are shown in Figures 13 and 14 as historical trends, based on the year the engine was certificated or qualified. It is seen that the higher power engines have better performance, and that engine performance overall has improved as a function of time as new technology features have been introduced. The technology improvements have largely been associated with materials and aerodynamic advancements, and we can reasonably expect such improvements at least to follow the trend projected from our history into the future.

But, it is believed that the historical projections are conservative. In the past, a major effort has been exerted to reduce engine complexity at the expense of performance. Reliability and initial cost cannot be sacrificed; however, with the new situation in fuel availability and increasing cost, engine manufacturers must take advantage of additional approaches to improve efficiency, including such things as the use of variable geometry and cooling schemes and recuperation. These will be discussed later.

Current transmission systems have reached a relatively high state of development. Allowable surface compressive stresses and gear root bending stresses have increased over 10 and 35 percent, respectively, over second generation design values. Initial times between overhaul (TBO) now start at a level to which earlier designs had to be gradually developed. The same is true for the power or torque per pound of transmission. Progress is being made in reducing transmission structure and airborne noise by use of such approaches as higher contact ratio gears. Today's transmissions offer more features--simple diagnostics in the form of chip light detectors, lower noise, and often the ability to continue operation for a significant time after loss of lubrication. This technology was originally spawned by military requirements and is known as "fly-dry." An alternative to a fly-dry transmission is to provide an emergency lubrication system. This, too, has been done successfully.

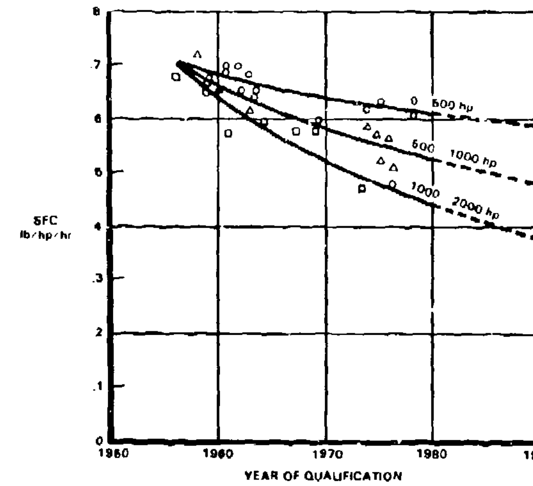


Figure 13 - Historical Trend of Turbine Engine Specific Fuel Consumption

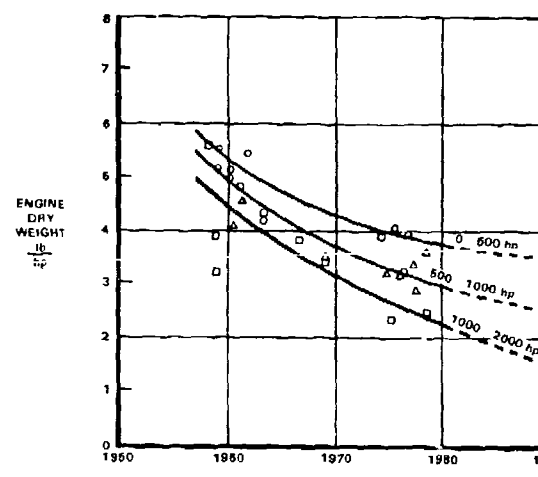


Figure 14 - Historical Trend of Turbine Engine Specific Weight

Power transfer couplings have improved significantly over the designs of two decades ago, and power matching in twin or triple engine designs is now an accomplished fact. New transmissions must keep pace with the technology level of new engines.

Current engines and the transmission systems that they drive are in a wide variety of configurations and technology levels. While all of them give, or promise to give, a creditable account of themselves in service, there is much room for improvement. The most significant areas that need attention include:

- Engine and transmission reliability
- Engine control system reliability
- Engine fuel consumption at cruise power
- Propulsion system initial and life-cycle costs
- Emergency power rating
- Engine and transmission size and weight
- Propulsion system health monitoring
- Noise and vibration levels
- Inlet air filtration
- Low grade fuel capability
- Engine/transmission/airframe integration

In the paragraphs that follow, the technologies needed in each of these areas will be discussed briefly, together with the promise for progress which they offer.

TECHNOLOGY ADVANCES NEEDED

Reliability

Once the helicopter is designed, its propulsion system selected, the development leading up to its qualification and/or certification is complete, and it is introduced into service, the early performance predictions tend to fade into the background, and the economics of its operation become uppermost in the owner's/operator's mind. Primary in this regard is the aircrafts' and their power plants' reliability; i.e., the ability to perform consistently in the manner required for the intended use.

NASA states[1] that approximately 49 percent of the instances of unreliability in modern turbine-powered helicopters occurs in the propulsion systems. Of these, 35 percent are attributed to the engine or its associated systems and 14 percent to the transmission. A review of helicopter accidents due to material failure shows that the engine and its systems cause over two-thirds, the transmission system less than 10 percent. Bell data indicate that in a typical helicopter fleet, an unscheduled interruption of operation due to aircraft material problems occurs once every 400 flight hours. About 30 percent of those interruptions are caused by the engine and transmission systems.

It is obvious that improvements that reduce the number of these interruptions will increase safety, reduce costs and increase customer satisfaction. In the great majority of cases, new technology is not required to improve reliability, merely a more careful attention to detail in the application of known principles. Early correction of the service-revealed difficulties is vital to the continued growth of the industry, and engine manufacturers must appreciate this.

Even though most cases of unreliability can be corrected easily, there are some areas where the current designs are being pushed to the limit, and future progress depends on the application of new technology.

For example, experience has shown that as many as two-thirds of the power interruptions in flight are caused by the present generation of partially open-loop, hydraulic- or pneumatic-mechanical engine control systems. Present-day and future gas turbines need more precise closed-loop scheduling of fuel flow and variable geometry features to match the pilot's power demands to the ambient conditions and engine requirements. The application of microelectronics to the fuel management and variable geometry systems, particularly during engine starting, shows promise of providing precise and reliable response of the engine to flight requirements and, thus, of reducing pilot workload, conserving engine life, and providing greater power reserves.

Operational recording devices and maintenance techniques can be developed that continuously monitor propulsion system health and total life used so as to give early warning of unfavorable operating trends and impending troubles. These systems will permit the establishment of least-cost maintenance programs that are consistent with the reliable operation of the propulsion system.

Eventually it may be possible to integrate the flight and engine controls for even greater simplification of the cockpit, reduced pilot workload and increased safety. It is here that a great part of the human error accidents that make up 70 percent of all accidents might be prevented. A word of caution is in order, however. The environment into which these devices are thrust must be clearly understood and protected against. In addition to heat and vibration, the ever increasing electromagnetic pollution of the atmosphere is of concern. There is a great deal to learn here, and we must proceed cautiously.

Transmission reliability might be most improved through better manufacturing techniques that provide ultra-high levels of consistency, thus eliminating the stresses associated with manufacturing variation. Improved inspection techniques are important so that inclusions can be minimized. Techniques for measuring root bending stresses should be more precise than observing tooth loading patterns.

In the design of propulsion systems, care must be taken not to overemphasize weight reduction, when a little more could provide significant improvement in reliability. The trend of contracting power-by-the-hour for the helicopter, as well as for the engine, can be expected to grow rapidly as our customers demand consistent, predefined cost of operation. This will cause design approaches to become more conservative.

An improved operating environment for the engine and transmission can contribute greatly to the reliability of the propulsion system. Inlet air filters or particle separators, crash-resistant suction fuel systems, highly effective fuel and oil filters and oil coolers, engine/airframe dynamic compatibility, vibration isolators and low-loss exhaust systems that minimize reingestion and protect the airframe from high temperatures are all important elements of successful helicopter propulsion systems. All require continuing improvement.

The integration of the engine and transmission is also quite important. Engine shaft speeds will always be pushed as high as aerodynamic refinements allow, so as to reduce the size of the engine. With higher engine output speeds, greater speed reduction is required. Thus the question, how much of this reduction should be taken as part of the engine, if any, and how much in the helicopter transmission? This question must be answered for each design, based on considerations of mounting and shafting. The possibility of mounting the engine and transmission together as a single unit should not be overlooked.

Specific Fuel Consumption

In the light of the current fuel shortage and ever increasing fuel costs, the fuel consumption of the engine at cruise power is an important factor in the earning power of the helicopter. The importance of fuel cost is indicated by Table I,[2] which shows the fuel costs for several Bell helicopters as a percent of direct operating costs (DOC) for various prices of fuel. In the future, even with the coming low-grade fuels, fuel costs can be expected to exceed the upper values shown, and, as they do, fuel consumption reduction becomes mandatory.

Helicopter Model No.	Fuel Cost, U.S. $/GAL			ASSUMPTIONS
	1.0	2.0	4.0	
	Fuel Cost As % DOC			
206B	16	25	36	· 1200 FLT HRS PER YEAR
206L-1	15	24	34	· DEPRECIATION @ 7% P.A.
205A-1	24	35	47	· INSURANCE @ 6% PURCHASE PRICE
212	24	36	49	· INFLATION @ 8% P.A.
222	16	25	36	· 5 YEAR INTERVAL FROM MIN TO MAX FUEL COSTS

Table I. Fuel Cost as a Percent of Direct Operating Costs

In the continuing evolution of gas turbine technology, reduction in specific fuel consumption can be achieved by higher pressure ratios and increased turbine operating temperatures, as well as by better compressor aerodynamics, improved combustion, more efficient turbines, and reductions of losses caused by tip clearance and friction.

In achieving these improvements, it is essential that an adequate stall margin be retained, so that the engine can operate smoothly throughout the flight regime. Engine performance deterioration must be minimized throughout the overhaul interval or provided for initially, and engine performance must be readily restored to original levels during overhaul. Inlet air cleaning and flight line washing techniques, together with good air filtration and simple, rugged designs, are contributory to this goal.

It is also most important that full power is available within a few seconds of pilot demand. This can most likely be achieved by better control of running clearances and temperature control systems. The requirement for almost instantaneous power response from the remaining engine(s) upon engine failure caus s one to look again at the single-shaft turbine or to consider the more complex and more efficient three-shaft designs. Power response time is extremely important.

Higher turbine operating temperatures will require more extensive use of turbine cooling for metal wheels, blades, and vanes combined with high-temperature coating or, alternatively, the introduction of ceramic components. This family of materials, with its high strength-to-weight ratio at elevated temperatures and corrosion resistance, shows promise of reliable high-temperature operation with increased useful lives and reduced weight and parts cost.

The use of a highly effective recuperative system to reduce fuel consumption is a possibility, especially for large, long-range rotorcraft or for military helicopters requiring infrared suppression. Even for smaller commercial helicopters, a recuperative system might be attractive if it were designed as an integral part of the fuselage to minimize it's size, weight, and drag impact.

Variable geometry and cooling schemes offer considerable promise. The variable cooling involves manifolding and throttling to reduce cooling air flow at part power operation for better efficiency. Variable geometry takes the form of variable compressor and turbine stators, variable exhaust and turbine nozzles, and gas-producer clutching arrangements. These devices do increase parts count and cost, but they offer the possibility of major improvements in efficiency. Table II[2] presents the probable specific fuel consumption and weight of the types of devices mentioned. It is seen that very significant improvements can be expected, and these add to the cycle and component efficiency increases projected from history.

Device	Percent Change	
	SFC Reduction	Engine Weight Increase
Variable Cooling	2	2
Recuperation	19	24
Variable Geometry with Recuperator	27	26
Variable Geometry without Recuperator	13	4

Table II. Potential of Engine Developments

Propulsion System Weight

Propulsion system weight is an important factor in the critical trade-off between engine plus fuel weight and useful load and range. Advanced aerodynamic technology can reduce the number of stages in the compressor and turbine, thereby reducing weight and increasing reliability.

Transmission gearing can be made somewhat lighter by refined design techniques, but presently known materials, heat treatments, surface processing, and lubricants are being pushed to the limits. The use of double-vacuum melt steels, new steels with higher allowable stress, improved lubricants, especially those with higher Ryder capacity, and silicon nitride bearings in special applications are possible avenues for reducing transmission weight. New configuration transmissions such as the use of traction drives, especially for the smaller systems, offer promise. Considerable weight savings are possible in the transmission area.

Figure 15 shows the relative Hertzian and root bending moment stress levels that have been used in transmission design in the past and similar data for new materials being evaluated now. Figure 16 shows the resulting increase in mean time between removal (MTBR). Clearly, time will bring further improvement here.

Composite materials are finding broader application in the propulsion system because of their high specific modulus, light weight, corrosion resistance, and low cost. These materials are now widely used in cowlings and inlet systems. High-temperature polyimide materials are extremely attractive in these applications. Although never used in production, successful drive system shafting has been made of graphite. Transmission cases have been made of composite materials. Figure 17 shows a composite transmission case in process for a Boeing Vertol CH47D helicopter.

We should pursue the promise of composites--including metal matrix composites--throughout the propulsion system. Possible applications include high-speed shafting, compressor components, and engine and transmission gear cases, providing the reduced heat transfer coefficient through them can be offset by increased cooler capacity with a net weight reduction. Development efforts should be continued to understand more fully and to control the unique properties of composites so that they can be used with the same confidence as homogeneous materials.

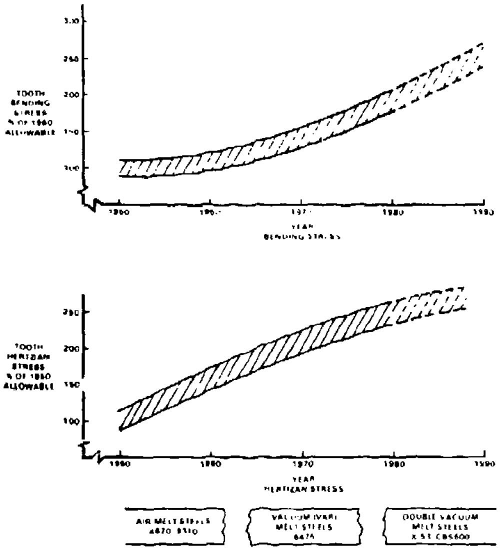


Figure 15 - Gear Strength Trend

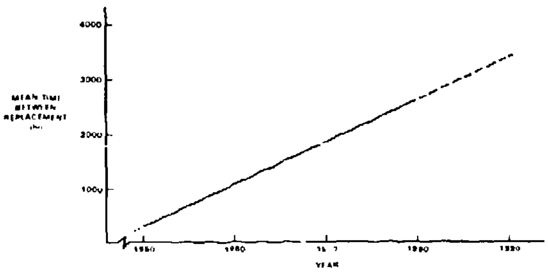


Figure 16 - Improvement in Gear Surface Failure with Time

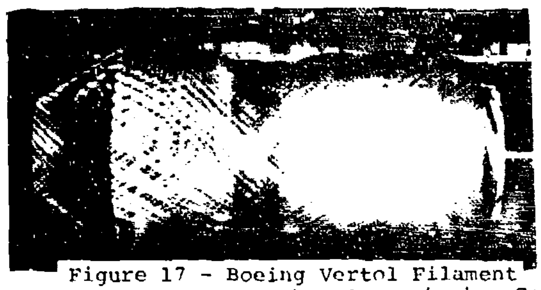


Figure 17 - Boeing Vertol Filament Wound Advanced Composite Transmission Case

Propulsion System Costs

Although propulsion system acquisition cost is a large part of the total cost of a helicopter, it is small when compared to the life-cycle cost of the system, which includes the fuel, lubricants, parts, maintenance, and labor used during the useful life of the system. Thus, it is prudent to consider life-cycle, cost-effective improvements to the helicopter and its systems, including the propulsion system, even though the acquisition costs may be increased.

The technologies discussed above aimed at increasing parts life and reliability and reducing fuel consumption must be viewed as they contribute to reducing life-cycle costs, rather than the shorter term acquisition costs. Similarly, propulsion system manufacturers must make the hard trade-offs such as between increased reliability and lower fuel consumption based on consideration of life-cycle cost. This concept will not be easy to sell to our prospective customers, unless it is backed up by innovative guarantees--and this is what we must do.

Rating Philosophy and Qualification/Certification Procedures

There is an urgent need for some truly innovative thinking on the part of industry and the civil regulatory agencies concerning the rating philosophy and certification procedures for helicopter engines, particularly those used in multiengined aircraft. In those applications, the maximum allowable gross weight of the helicopter is not limited by the total installed power, but rather by the amount of power available from the remaining engine(s), should one engine fail at the most critical point during takeoff.

This leads to a set of one-engine-inoperative (OEI) ratings for use during emergency conditions, which occur very infrequently, if at all, during the life of a given engine or helicopter. At present, the highest emergency rating in current engines is approximately 110 percent of the takeoff rating and is called a 2-1/2-minute rating. It is demonstrated by fifty 2-1/2-minute runs for a total of 2 hours and 5 minutes of testing in the United States Federal Aviation Administration's required 150-hour certification endurance test. By contrast, the same test schedule requires only 11 hours, 40 minutes of operation at takeoff power, which is used about once every flight. At the conclusion of the test, the inspection of the engine parts must not reveal any impending failure that would compromise further safe operation.

The infrequent situations requiring emergency power during takeoff would normally exist for no more than about 10 to 20 seconds of operation of the surviving engine(s) and would occur only to complete a takeoff after one engine failure at the critical altitude and temperature condition, with no wind and with no suitable landing area available. This situation does not occur often, and when it does occur, the required duration of the emergency power is very low.

The amount of testing at the 2-1/2-minute rating is disproportionately high. Because of the long test time, the emergency power level that can be certificated is limited due to the creep and stress rupture considerations of the turbine rotor. If a shorter demonstration period were acceptable--say ten 30-second bursts to a maximum emergency power during the 150-hour test--the rating could be as much as 125 percent of the takeoff rating, thus increasing the allowable payload of the helicopter by approximately 50 percent.

Even higher OEI ratings, perhaps as much as 150 percent, would be possible as a "burnout" rating, demonstrated by a single 30-second burst followed by 15 minutes at normal rated power (NRP) on the type test engine following reassembly after the post-test inspection. The burnout rating would be certificated with the understanding that that rating would be used once, and upon its use, the engine would be returned for overhaul.

There is also the need for a higher enroute emergency OEI power rating. Although this rating should be of a much longer time duration, the approach of requiring engine overhaul after the use of that power would allow the definition of an acceptable certification test. The time duration for the enroute emergency OEI power rating should allow time for the disabled rotorcraft to fly to a location where low altitude flight could be maintained at NRP. Although this will vary depending on what area of the world the flight takes place, it is believed that a one-hour emergency enroute power rating would suffice, with flight at NRP allowed for an additional two hours.

The aerodynamic design of the compressor and its operating line may have to be adjusted to achieve these higher OEI ratings without surge, possibly resulting in a slight compromise in cruise fuel consumption. Even so, there would be a reduction in total fuel used for a given payload because of the use of higher percentage of NRP during cruise.

THE CHALLENGE

It is visualized that future helicopter propulsion systems will consist of two or more turboshaft engines driving through high reduction ratio, lightweight, on-condition, fail-safe transmissions. The engines will have a higher cycle pressure ratio, more efficient compressors and turbines, operate at higher turbine gas temperatures than present engines and incorporate variable geometry features. These engines will be free from surge and capable of extremely rapid power response under all conditions.

The control systems would be the full authority, electronic type with engine and transmission health monitoring and outputs to the flight control system. Both the basic engine and transmission will be more compact and lighter in weight per horsepower produced or transmitted, and the engines will use considerably less fuel at cruise power than they currently do. With these new engines, safety and economy will be provided through the application of rational OEI emergency power ratings. All of these gains will be accompanied by concurrent improvement in system reliability and a reduced life-cycle cost.

The gross weight of a helicopter that is powered by this advanced propulsion system will be about 20 percent less than today's helicopter doing the same job, and it will use at least 25 percent less fuel and travel at about a 10 percent higher speed--because of its improved propulsion system.

THE CHALLENGE TO THE PROPULSION SYSTEM SPECIALIST IS TO IDENTIFY THE TECHNOLOGIES REQUIRED TO ACCOMPLISH THESE GOALS, TO LAY PLANS TO BRING THEM ABOUT, AND TO MAKE IT HAPPEN.

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1. Dougherty, John J. III and Lawrence D. Barrett, RESEARCH REQUIREMENTS TO IMPROVE RELIABILITY OF CIVIL HELICOPTERS, NASA CR-145335, NASA Langley Research Center, April 1978.

2. Lynn, Robert R., REQUIREMENTS AND POSSIBILITIES, ROTORCRAFT POWER PLANTS, a lecture to the Hatfield Branch of the Royal Aeronautic Society, 11 March 1980, and published in The Aeronautical Journal, Volume 85 Number 840, January 1981.

DISCUSSION

W.Crawford, US

I question the capability to achieve 50 percent emergency power capability without burn-out. Modern engines have temperature/power derivatives that will not give that much additional power for significant temperature increases.

Author's Reply

Twenty-five percent should be available for present engines. Fifty percent is believed to be achievable through the use of ceramic materials in critical locations.

K.Rosen, US

What plan would you offer for emergency power certification?

Author's Reply

Joint approach to the certifying agencies by the engine manufacturers, helicopter manufacturers, and operators, all acting in concert. Actual time histories of simulated or real engine failures at critical points in the flight path would be a start.

P.Brammer, UK

Is a 30-second burst at emergency power adequate?

Author's Reply

Perhaps not. This value should be settled by pilots. Some of them have indicated ten seconds is enough but it is hoped that the number would come out between 30 and 45 seconds.

PROGRAMMES D'ESSAIS DE DEVELOPPEMENT ADAPTES AUX MOTEURS D'HELICOPTERES

par

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1 - INTRODUCTION

Dans un programme général de développement d'un moteur, c'est à dire depuis la première rotation au banc jusqu'à la mise en service du modèle développé en passant par la mise au point du modèle de base, le chemin à parcourir est semé de contraintes de plusieurs natures :
- en premier lieu des exigences règlementaires qui sont constituées par les règlements de Navigabilité pour les moteurs civils, par des Spécifications techniques particulières pour les moteurs militaires ;
- en second lieu des exigences d'ordre opérationnel qui viennent se juxtaposer aux précédentes et qui sont constituées :
  . par les divers types d'utilisation envisagés : profils de vol des différentes missions.
  . par les conditions d'environnement susceptibles d'être rencontrées.

Ce second volet est particulièrement important pour un turbomoteur d'hélicoptère du fait de la variété des utilisations et des conditions d'environnement comparées à celles d'un réacteur ou d'un turbopropulseur.

Pour répondre à l'ensemble de ces exigences il faut bâtir, entre autres, un programme d'essais qui prenne en compte ce faisceau de contraintes, parfois contradictoires, d'une part pour obtenir la Certification et d'autre part pour effectuer la mise en service, l'exploitation et le développement sans aléas majeurs.

Il peut paraitre ambitieux d'aborder un sujet aussi vaste dans un aussi court laps de temps, aussi nous nous limiterons à examiner quelques essais spécifiques à un turbomoteur d'hélicoptère.

2 - PROGRAMME MAKILA

Nous avons choisi comme exemple le programme du turbomoteur MAKILA qui équipe l'hélicoptère AEROSPATIALE AS-332 Super PUMA.

Rappelons brièvement l'origine et les principales étapes du programme MAKILA. Lancé en 1974 pour répondre aux besoins du marché des hélicoptères de moyen tonnage à long rayon d'action, le MAKILA turbomoteur de la classe des 1300 kW (1800 ch) est d'une conception moderne, entièrement modulaire tout en possédant l'architecture générale des moteurs TURBOMECA.

Il développe en conditions Standard au niveau de la mer 1240 kW aux régimes de décollage et intermédiaire d'urgence (ou 30 minutes) et 1310 kW au régime maxi d'urgence (ou 2 1/2 minutes), la puissance thermique maximale est de 1 400 kW.

Etapes clés du programme

première rotation au banc en réacteur	Novembre 1976
premier vol sur PUMA SA 330	Juin 1977
premier vol sur Super PUMA AS 332	Septembre 1977
Réception des premiers moteurs de série	Décembre 1979
Certification DGAC	Février 1980

3 - EXIGENCES REGLEMENTAIRES DE NAVIGABILITE

Comme pour tous les turbomoteurs TURBOMECA pour les hélicoptères qui ont à la fois des applications civiles et militaires, nous avons choisi d'emblée de répondre à l'enveloppe des règlements civils et à des points particuliers des Spécifications militaires pour les moteurs, notamment en ce qui concerne les essais d'environnement.

Le MAKILA répond donc à la fois aux exigences des règlements de navigabilité suivants :

- Le JAR-E ou Joint Airworthiness Requirements-Engines à l'édition 3 qui est le règlement européen pour les moteurs, basé sur le règlement Britannique B.C.A.R Section C à l'édition 10. Pour information, le JAR-E n'étant applicable aujourd'hui qu'aux moteurs d'avions il a été adapté pour pouvoir correspondre à l'installation motrice définie par la FAR 29, notamment en ce qui concerne les régimes d'utilisation.

- La FAR Part 33 à l'amendement 8

Le MAKILA a également subi des essais définis par les Spécifications pour les moteurs militaires MIL-E-8593 A et D.Eng RD 2100.

L'ensemble de ces règlements constitue la base du programme des essais de validation du moteur.

Par rapport aux autres types de moteur ce qui caractérise essentiellement les moteurs d'hélicoptère du point de vue des règlements, c'est l'existence de regimes d'urgence pour les appareils multimoteurs classés en Catégorie A c'est à dire certifiés pour assurer un transport public de passagers :

- Régime maximal d'urgence selon définition BCAR Section C ou régime 2 1/2 minutes selon définition FAR Part 1 destiné à couvrir le cas de panne d'un moteur au décollage pour les hélicoptères.
  Ceci signifie qu'en cas de panne d'un moteur, un surcroit de puissance doit être instantanément disponible sur le ou les autres moteurs pour permettre à l'hélicoptère de poursuivre la phase de décollage.

- Régime intermédiaire d'urgence à durée illimitée selon définition BCAR ou Régime 30 minutes selon définition FAR Part 1

L'existence de régimes d'urgence constitue une donnée fondamentale pour la conception du moteur mais aussi pour le programme d'essais. Tel est le cas pour le MAKILA. Nous verrons dans la suite de l'exposé quelles sont les implications particulières à ces régimes sur les démonstrations apportées.

4 - EXIGENCES OPERATIONNELLES

Les contraintes issues des exigences opérationnelles viennent s'ajouter aux exigences règlementaires. Elles forment l'essentiel de la particularité des turbomoteurs d'hélicoptères par rapport aux autres types de moteur. Elles sont constituées par la variété des types d'utilisation alliée à un large éventail de conditions d'environnement.

La difficulté réside dans la simulation de ces conditions dans un programme d'essais nécessairement contracté dans le temps afin d'anticiper au mieux les conditions réelles d'utilisation.

Les différents facteurs dont il faudra tenir compte sont en particulier :

. le type de mission

- transport :
  . transport public ou privé de personnel
  . liaison avec les plate-formes de forage en mer, acheminement de personnel, de matériel.
  . transport et secours en montagne
  . liaison aéroports/villes

- travail aérien
  . transport de charges à l'élingue
  . grutage
- surveillance de lignes électriques à haute tension, de routes, de pipe-lines etc...
- évacuation sanitaire
- missions militaires diverses

. l'environnement
les différents types de mission énumérés sont réalisés dans des environnements très divers tels que :
- atmosphère saline propice à la corrosion (notamment pour la desserte des plate-formes en mer)
- atmosphère sableuse propice à l'érosion principalement dans les régions du Moyen Orient
- atmosphère chaude et humide des pays tropicaux également propice à la corrosion ainsi qu'au développement des micro-organismes dans le carburant
- atmosphère polluée des régions fortement industrialisées pour les missions près du sol de survol de lignes haute tension ou de routes
- atmosphère froide et vol sous la neige
- vol sous pluie intense en pays tropical

. les opérations d'entretien
la qualité des opérations d'entretien varie considérablement du fait de la multiplicité des utilisateurs d'hélicoptères, de leurs moyens, de la dispersion géographique et en particulier de l'éloignement des bases d'entretien des zones d'utilisation.

L'exposé de ces exigences montre bien l'extrême variété des conditions règlementaires et opérationnelles qu'il faut couvrir. Mais en outre, si des équipements optionnels permettent à l'hélicoptère de s'adapter à des travaux et des environnements très différents, le turbomoteur, lui, restant en place, doit assurer cette polyvalence dans tous les cas.

Les différentes contraintes exposées ci-avant interviennent en effet soit isolément, soit combinées et les essais devront donc en constituer une enveloppe représentative.

On imagine aisément que des choix doivent être faits pour les programmes d'essai car il n'est pas possible de reproduire exactement au banc toutes les conditions susceptibles d'être rencontrées en service.

Il est donc nécessaire d'effectuer des essais de simulation au banc sur éléments séparés et sur moteur complet pour s'approcher au plus près des conditions opérationnelles et anticiper aussitôt que possible les problèmes qui risquent de se poser en utilisation.

5 - PROGRAMME D'ESSAIS MAKILA

Dans l'ensemble du programme d'essais du turbomoteur MAKILA nous avons choisi trois essais qui ont pour objet de simuler quelques unes des conditions exposées précédemment et qui sont particulières à l'utilisation d'un hélicoptère.

Il s'agit d'essais qui se rapportent aux questions suivantes :
. Temps de réponse en puissance et transfert des charges en cas de panne d'un moteur
. Tenue du moteur en longue endurance
. Résistance du moteur à la corrosion

5.1. Temps de réponse en puissance

Sur hélicoptère bi-moteur une des conditions importantes que doit remplir l'installation motrice, c'est comme nous l'avons mentionné précédemment la continuation du vol en cas de panne d'un moteur, même pendant la phase de décollage.

Cette condition implique de réaliser une régulation qui assure les fonctions suivantes :
- transfert instantané de charge des 2 moteurs sur un seul
- puissance maximale disponible dans un délai très court (régime maximal d'urgence ou 2 1/2 minutes) pour des raisons évidentes de sécurité
- performances en transitoire permettant cette puissance sans pompage, sans instabilité et sans dépassement des limites autorisées de vitesse, de couple et de température des gaz.
- précision d'affichage de cette puissance de façon à éviter les marges excessives préjudiciables à l'évaluation des performances de l'hélicoptère.
- fidélité d'affichage de cette puissance à la fois pour des raisons de sécurité et pour éviter le dépassement des limites de fonctionnement du moteur.

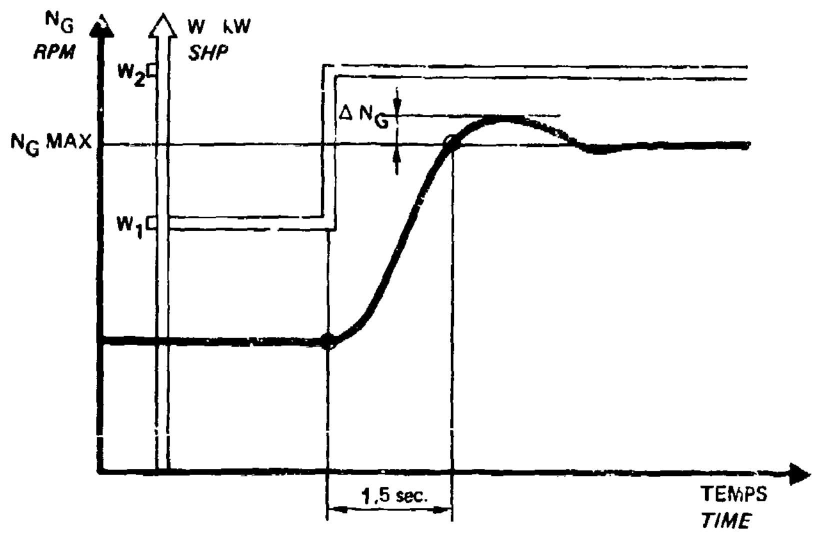
C'est l'association d'une régulation de carburant de qualité ayant des marges confortables avant décrochage des compresseurs avec une bonne combustion en regime transitoire qui permet de remplir ces fonctions.

Sur le MAKILA la solution choisie pour la régulation consiste en un régulateur mixte hydromécanique et électronique qui maintient la vitesse de turbine libre constante par action sur le débit carburant avec une fonction d'anticipation grâce à une liaison avec le pas général de l'hélicoptère.

Nous ne rentrerons pas ici dans le principe ou dans le détail de la construction du régulateur, ce n'est pas l'objet de l'exposé. Nous allons plutôt examiner les essais à mettre en oeuvre pour vérifier que toutes les fonctions énumérées précédemment sont réalisées d'abord sur le régulateur seul, ensuite sur le moteur au banc puis sur le moteur en vol et enfin comment les différents paramètres spécifiques à l'utilisation interviennent.

a - Essais de simulation sur régulateur

Les premiers essais à mettre en oeuvre sont effectués sur le régulateur seul sur un banc de simulation.



A partir d'un fonctionnement stabilisé correspondant à une croisière à vitesse donnée par exemple, on a un état de charge sur le moteur qui correspond à une vitesse du générateur de gaz Ng, image de la puissance soit W_1. On réalise alors instantanément un échelon de puissance par une simulation de charge sur le régulateur à pas général constant jusqu'à une valeur correspondante de puissance W_2 qui représente la puissance maximale d'urgence, c'est à dire la panne d'un moteur. On s'est fixé préalablement le temps minimum pour obtenir cette puissance, lequel représente le temps mini de restauration de la puissance pour que le pilote puisse maintenir l'hélicoptère en vol dans la phase la plus critique.

On vérifie alors sur le régulateur que ce temps fixé à 1,5 sec. pour le MAKILA n'est pas dépassé et on contrôle en même temps que le N_g s'établit à la valeur maximale prédéterminée sans que le ΔNg soit supérieur aux limites validées, et sans oscillation du régulateur.

On réalise ainsi grâce au simulateur le contrôle :

- de la disponibilité de la puissance maximale
- de l'affichage de cette puissance sans instabilité
- de la fidélité et de la précision de cet affichage
- de la rapidité d'obtention de la puissance maximale

Ces mêmes essais sont ensuite réalisés en simulant toutes les conditions d'environnement envisagées :

. température de carburant fonction du domaine de température ambiante prévue c'est à dire de -50°C à + 50°C. En fait le domaine à considérer s'étend de -50°C à + 70°C pour tenir compte des températures d'imprégnation des réservoirs et des tuyauteries d'un hélicoptère restant plusieurs heures sur un parking où la température extérieure est de + 50°C.

. pression atmosphérique c'est à dire altitude-pression dont l'influence se fait sentir sur un dispositif de contrôle d'accélération qui élabore une loi de débit.

Les essais au banc consistent à établir cette loi pour un fonctionnement au sol de sorte qu'elle couvre l'ensemble du domaine en altitude. Les vérifications sont faites ultérieurement lors des essais en vol.

. conditions d'alimentation du moteur en carburant, c'est à dire les conditions de pression ou de dépression à l'aspiration de la pompe HP.
On fait donc varier pour cela le débit et la pression à l'entrée pour simuler le fonctionnement avec et sans pompe de gavage et on vérifie que l'on obtient le Ng maxi dans ces conditions.

. type de carburant

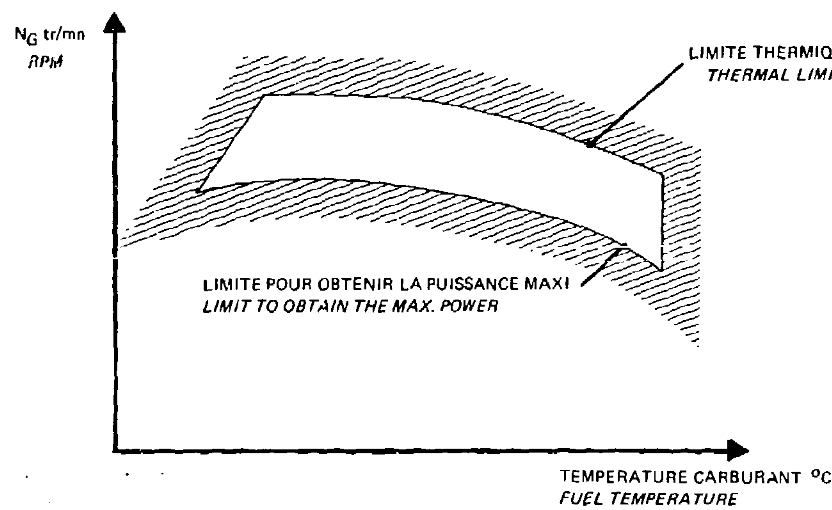
La variété des carburants à utiliser est là aussi une des caractéristiques de l'utilisation des turbomoteurs d'hélicoptères du fait, bien entendu, de la multiplicité des utilisateurs, des régions et des disponibilités : depuis les carburants approuvés pour l'utilisation normale Jet A1, JP8, JP4 et JP5 jusqu'aux carburants de remplacements avec restrictions d'emploi tels que le gaz-oil routier ou l'essence aviation.

Les essais décrits ci-dessus sont donc effectués successivement avec les divers carburants prévus en utilisation pour vérifier les performances du régulateur au banc et définir les éventuelles limitations d'emploi.

b - Essais sur moteur au banc

Lorsque la mise au point du régulateur au banc de simulation est terminée il faut vérifier cette fois sur moteur complet que toutes les fonctions sont bien réalisées avec la charge représentée par un frein sur l'arbre de sortie moteur. La différence essentielle avec l'installation sur hélicoptère est l'inertie entre le frein du banc et la transmission jusqu'au rotor.

Du fait de l'inertie et de la régulation de charge du frein la fonction temps d'établissement de la puissance maximale ne peut être contrôlée au banc. On contrôle par contre la valeur de cette puissance, c'est à dire en fait la valeur du Ng maxi en augmentant progressivement la charge du frein.



On vérifie ainsi que la valeur de Ng se trouve entre les limites suivantes correspondant à l'ensemble du domaine de fonctionnement :

- limite thermique maximale validée pour le moteur
- limite pour obtenir la puissance maximale

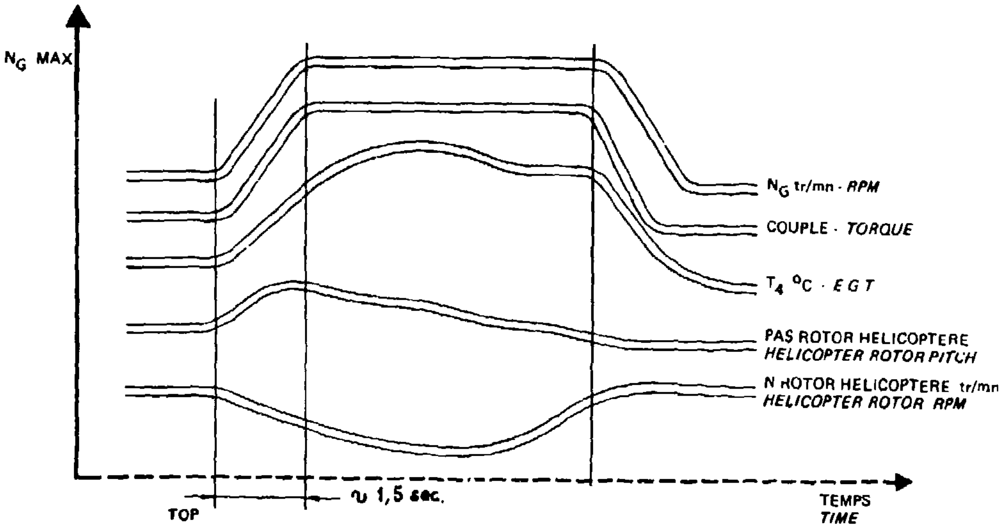
c - Essais sur moteur en vol

Tous les essais préliminaires étant effectués on considère que toutes les garanties sont prises pour obtenir en vol la puissance maximale d'urgence mais il reste à le vérifier et, en particulier, à déterminer quelle est l'influence de l'inertie de la transmission et du rotor de l'hélicoptère sur la réponse du régulateur, ce qui n'avait pu être fait au banc.

Cette vérification fait l'objet d'une campagne d'essais sur hélicoptère où l'on réalise la manipulation suivante : à pas constant en montée en bi-moteur on ramène brutalement la manette de commande du débit de carburant d'un des moteurs de la position "régulation" à la position "ralenti-vol" correspondant à un cran et l'on observe l'évolution des paramètres : N_G du 2e moteur et N rotor principalement.

ESSAIS SUR MOTEUR EN VOL

ENGINE FLIGHT TESTS



L'évolution du N_G permet donc finalement de s'assurer que le transfert de puissance s'effectue bien sur le moteur restant et qu'il délivre bien la puissance nécessaire à la poursuite du vol dans un temps compatible avec les conditions établies sans oscillation ni dépassement des limites approuvées.

Cette même vérification doit ensuite être menée sur un nombre significatif de régulateurs et dans toutes les conditions revendiquées à savoir :

- altitude
- différents types de carburant
- carburant chaud et froid
- panne de pompe de gavage

Ce parcours complet étant achevé, on est assuré d'avoir couvert tous les types d'utilisation envisagés avec le maximum de garanties.

5.2. <u>Essai d'endurance en cycles de vol-type</u>

Le deuxième exemple toujours tiré du programme de développement MAKILA concerne l'endurance du moteur en cycles que nous avons convenu de désigner "vol-type".

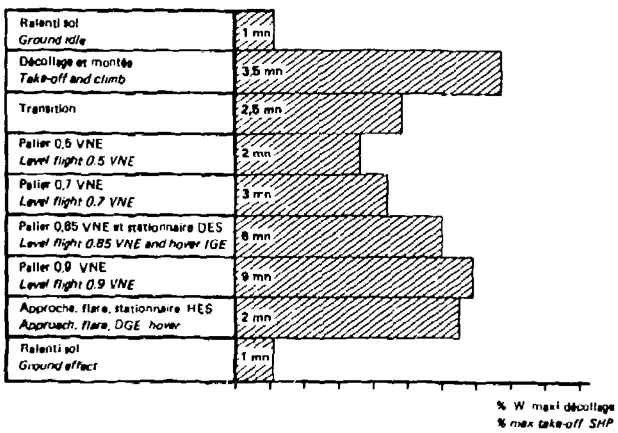
Comme nous l'avons vu précedemment la variété importante des utilisations de l'hélicoptère rend difficile la détermination d'un seul et même profil de mission. Il est donc nécessaire de faire certaines hypothèses simplicatrices. En fait le profil de vol-type quel qu'il soit constitue une référence à partir de laquelle il est possible de déterminer les influences de tel profil plus ou moins sévère.

La base de départ est constituée par le ou les spectres d'utilisation de l'hélicoptère exprimé en puissance totale à l'entrée BTP et calculé pour la masse maximale en conditions Standard. L'exemple suivant est donné pour la simulation d'une utilisation civile de transport. D'autres simulations peuvent exister pour des utilisations plus sévères, en particulier militaires.

Ce spectre est synthétisé et ramené en puissance par moteur avec les temps aux différents paliers.

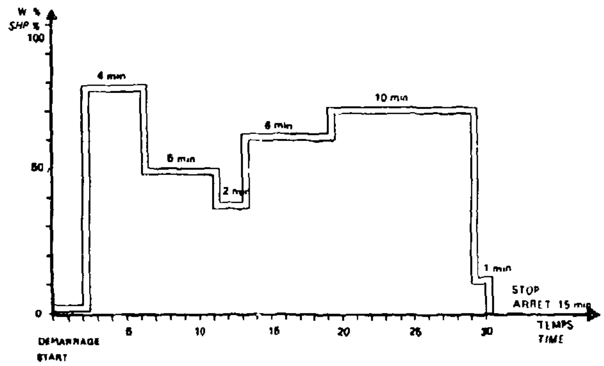
SPECTRE TYPE TRANSPORT CIVIL

CIVIL TRANSPORT TYPE SPECTRUM



PROFIL DU CYCLE D'ESSAI

TEST CYCLE PROFILE



On en déduit un cycle type pour le moteur dont la durée totale a été choisie à 30 minutes. Ce cycle de référence correspond aux Conditions Standard au niveau de la mer et il faut donc simuler l'utilisation du moteur à des conditions thermiques plus sévères. A cet effet une partie des cycles est réalisée à Standard + 20°C et une autre partie à Standard + 35°C correspondant à la limite du domaine de fonctionnement pour le moteur. 4 000 cycles de 30 minutes, c'est à dire 2 000 heures d'essai ont ainsi été réalisées. La simulation du fonctionnement à température ambiante élevée est réalisée d'une part en augmentant la vitesse du générateur de gaz pour obtenir la puissance correspondante au Std et d'autre part en effectuant un prélèvement d'air derrière le compresseur centrifuge de façon à obtenir la température d'entrée turbine correspondant au fonctionnement à Standard + 20°C et + 35°C.

L'ensemble de ces dispositions doit faire l'objet de l'accord des Services Officiels français et d'ailleurs ce type d'essai est réalisé la plupart du temps au Centre d'Essais des Propulseurs à Saclay sous la surveillance directe de ces Services, ce qui fut le cas pour le MAKILA.

Le but principal de cet essai étant de vérifier l'endurance des éléments du moteur pour le potentiel envisagé des démontages réguliers sont effectués pour contrôler l'état des pièces. On profite également de cet essai pour mettre au point et valider les méthodes d'entretien : inspections périodiques par endoscope, lavage compresseur etc...

5.3. Essai d'ingestion d'eau salée et de susceptibilité à la corrosion

Une des utilisations les plus importantes de l'hélicoptère sur le plan civil est le transport "off-shore" pour les liaisons avec les plate-formes de forage en mer.

Un turbomoteur d'hélicoptère doit donc être capable de résister à l'agressivité de l'atmosphère saline sans pour autant avoir des caractéristiques technologiques ou de construction spécifiques à l'utilisation marine. En effet comme nous l'avons vu précédemment la polyvalence des hélicoptères nécessite que le turbomoteur puisse s'adapter à toutes les conditions d'environnement envisagées et il est donc prévu dans le programme général des essais d'ingestion d'eau salée et de susceptibilité à la corrosion.

Ces essais sont empruntés aux Spécifications pour les moteurs militaires Britanniques D.Eng-RD 2100 ou Américains MIL-E-8593-A destinés à couvrir les utilisations marines.

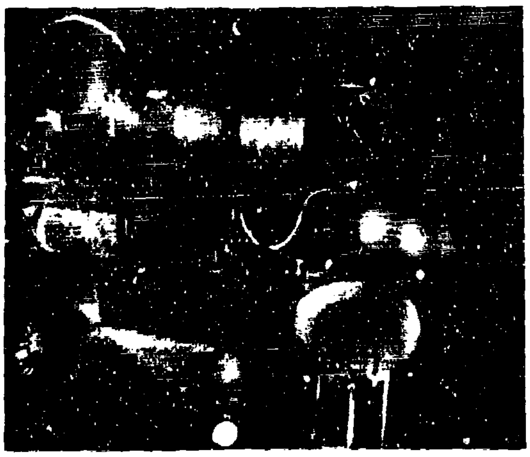
5.3.1. Essai d'ingestion d'eau salée

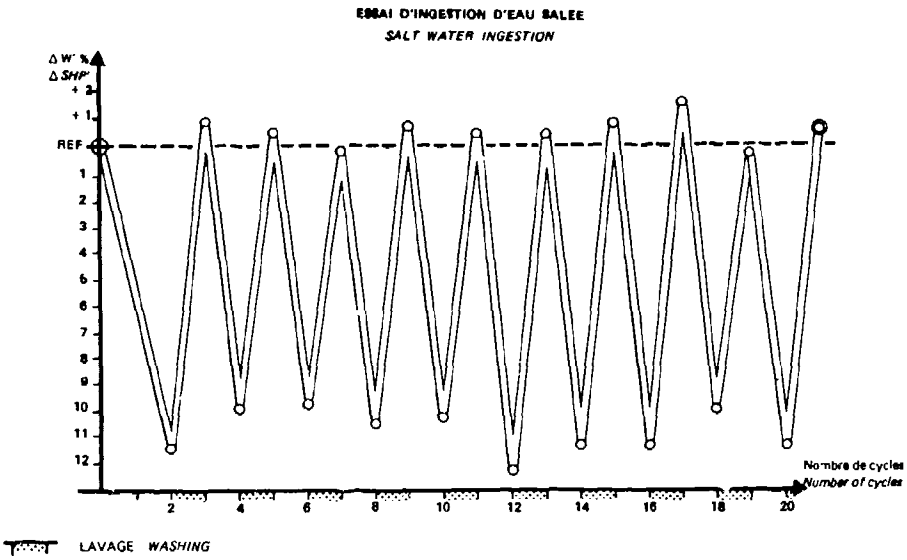
Cycle

PHASE	DUREE EN MINUTES		FONCTIONNEMENT	INJECTION D'EAU SALEE
	Partielle	Cumulée		
1	5	-	Régime Puissance Décollage	Marche
2	20	25	Régime Puissance Maximale continue	Marche
3	5	30	Ralenti	Arrêt
4	-	-	2 accélérations du ralenti au régime décollage	Arrêt
5	-	-	Contrôle des performances	Arrêt

Le débit d'eau injecté a été réglé à 0,6 l/h, il est réparti uniformément dans l'entrée d'air au moyen d'un injecteur calibré.

VUE GENERALE
DE L'INSTALLATION





20 cycles de 30 minutes, soit 10 heures d'essai sont ainsi réalisées. Ces essais sont mis à profit à la fois pour vérifier le comportement des organes internes du moteur à la corrosion par le sel et pour "mesurer" l'efficacité des méthodes de lavage du moteur.

Deux lavages avaient été prévus tous les 2 cycles avec un mélange d'eau déminéralisée et d'un inhibiteur de corrosion à 2%. Le lavage s'effectue moteur en fonctionnement au ralenti pendant 5 minutes, la quantité injectée est d'environ 5 litres.

Après le lavage, le moteur est protégé par injection dans l'entrée d'air pendant l'auto-rotation d'un produit hydrofuge pur.

Un relevé des performances est effectué au début et à la fin de chaque cycle et si l'on observe l'évolution des paramètres débit d'air, rapport de pression, température d'entrée turbine, puissance et consommation on peut :
- mesurer la perte éventuelle de performances
- déterminer les organes du moteur qui se dégradent
- évaluer l'efficacité de la méthode et du produit de lavage

On observe que la perte de performances due à l'encrassement par le sel est importante, jusqu'à 12,5% sur la puissance mais que la lavage effectué permet de restaurer les performances initiales (et même plus du fait que le moteur d'essai n'était pas "neuf").

Sur le MAKILA ces essais ont permis de mettre au point notamment une rampe de pulvérisation installée à poste fixe sur l'hélicoptère, l'injection du produit de lavage pouvant même être commandée depuis le poste pilote.

5.3.2. Essai de susceptibilité à la corrosion

L'essai se déroule de la façon suivante :

a) Protection interne et externe du moteur :
- injection en ventilation de 0,5 l d'un produit hydrofuge à l'intérieur du moteur
- pulvérisation externe de 0,6 l de ce même produit

b) 2 heures après la protection injection d'eau salée en ventilation jusqu'à ce qu'un brouillard épais sorte en continu pendant 1 minute au moins par la tuyère

c) pulvérisation externe du moteur avec la solution d'eau salée

d) obturation des orifices d'entrée d'air, de tuyère et de vanne de décharge

e) le moteur reste ainsi imprégné pendant une semaine puis on recommence la même opération sans protection préalable et on laisse à nouveau l'eau salée agir pendant une semaine

f) à la fin de la seconde semaine le moteur est démonté pour établir l'étendue et l'emplacement des zones corrodées.

Cet essai très sévère permet de préciser après analyse quelles sont les modifications à apporter pour obtenir une protection efficace.

On observe par exemple les zones où le sel s'est déposé sans corrosion et les zones telles que certains plans de joints qui nécessitent une protection plus efficace.

6 - CONCLUSION

Les contraintes de différentes natures qui sont imposées au constructeur d'un moteur d'hélicoptère pour répondre aux exigences tant règlementaires qu'opérationnelles nécessitent d'établir un programme d'essai qui s'approche au plus près de l'utilisation.

Nous avons essayé dans cet exposé de montrer par quelques exemples tirés du programme MAKILA comment ces exigences se traduisaient dans la pratique des essais.

C'est en restant attentif aux besoins des utilisateurs d'hélicoptères que le constructeur peut mettre en service des moteurs toujours mieux adaptés au marché, toujours plus performants, fiables et économiques.

AIRCRAFT TURBINE ENGINE DEVELOPMENT - CURRENT PRACTICES AND NEW PRIORITIES

by

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SUMMARY

The T700 engine program was conducted during the 1970s and is therefore representative of recent practices employed in the development of turboshaft engines for U.S. military application. The engine, which is in the 1,600 horsepower class, recently entered service in the twin-engine U.S. Army UH-60A Black Hawk helicopter. The T700's field introduction follows an extensive program of technology demonstration, development, qualification and maturity, which was conducted by the General Electric Company under contract with the United States Army. The paper surveys requirements applied in the T700 program and associated benefits, challenges and penalties. Suggested improvements for future programs are offered, and technology needs revealed during T700 development are identified. Post-qualification maturity testing, which was conducted to provide early exposure of high-time failure modes, is described. Program features which are important for maximum development costs payback are summarized.

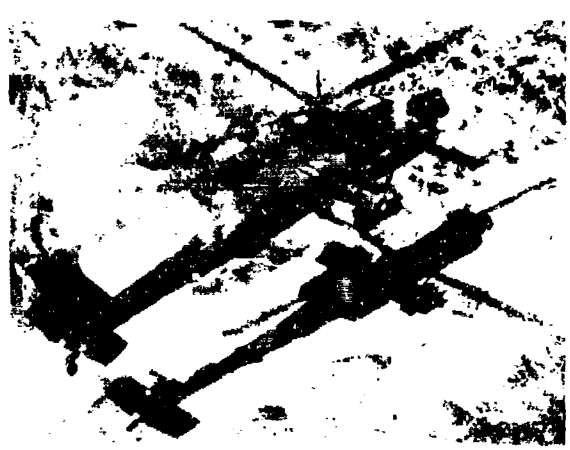
SECTION I

PROGRAM REQUIREMENTS

by Charles C. Crawford, Jr.

This section describes major development and qualification requirements which had their first U.S. Army application in the T700 program. General turboprop specifications from the 1950s were the starting point in developing these requirements. The specifications were updated with requirements subsequently generated for new U.S. Air Force and U.S. Navy engine programs, and incorporated unique Army requirements based on then current helicopter operational experience. The resulting document was included in the Army's request to industry in 1971 for an engine to power the planned Utility Tactical Transport Aircraft System (UTTAS).

Under the performance requirements, 30-minute Intermediate Rated Power (IRP) and Maximum Continuous Power (MCP) ratings were specified. This approach, plus the use of maximum continuous limits for the drive system, would virtually eliminate the pilot's concern for exceeding time-at-power limitations. At sea level standard day conditions, minimum IRP of 1,500 horsepower was required. A minimum acceptable level of power to be available at 4,000 feet/95°F (1219M/35°C) was also specified to ensure that the UTTAS could perform its primary mission under altitude and hot day conditions. The U.S. Army considers that the 4,000 feet/95°F capability will permit operation over 95% of the world. Maximum allowable specific fuel consumption at 60% power was defined to ensure selection of engine cycle parameters which would minimize fuel consumption at helicopter cruise conditions. These requirements, plus maximum allowed engine weight, established the turbine inlet temperature, pressure ratio and airflow. The specifications required extensive altitude engine testing to verify performance guarantees throughout the operating envelope. Minimum engine performance is documented by a computer program which is used to define aircraft operator's manual performance data and to generate power available data when considering new engine applications.



Hughes AH-64 AAH on Simulated N-O-E Mission

Regarding the control systems, the UTTAS engine specification contained several requirements intended to reduce pilot workload and to automatically provide overspeed and overtemperature protection. These features allow the pilot to devote maximum attention to mission accomplishment and to minimizing vulnerability. Constant power turbine speed governing requirements allowed no more than 3% speed variation during transients and no more than 1% during steady state operation. Torque matching requirements specified a capability to automatically match the output shaft torque of all engines in a multi-engine installation within 5% of the torque available from a single engine at MCP. In addition to primary control speed limiting, a secondary system was required to automatically limit power turbine overspeed in

the event of total loss of load (output shaft failure, etc.). Direct turbine gas temperature limiting was also required.

Cold weather operation with earlier generation helicopters revealed the need for more stringent cold starting requirements. As a result, the UTTAS engine specification called for starting with -65°F (-54°C) JP-4 fuel and with 12 centistoke (12 mm^2/s) JP-5 fuel. A cold temperature starting and acceleration test was specified which required four successive starts at each of these conditions. The starts were to be preceded by a ten-hour cold soak after the main bearings had reached test temperature. The JP-5 requirement also provides an improved JP-8 cold starting capability relative to older U.S. Army engines.

The UTTAS engine requirements also included new engine life criteria. For example, 5,000-hour design life to a power spectrum which specified 15% time at IRP was required. The specification called for increased endurance testing; two 150-hour qualification tests, with 50% time at IRP. Hence a thorough evaluation of hot section stress rupture life could be obtained. Tightened allowable performance degradation requirements were introduced to reduce performance-loss damage such as turbine airfoil high temperature erosion and seal wear. The endurance tests also incorporated a requirement to verify stable operation with a wide range of aircraft rotor systems. Maximum and minimum allowable power absorber polar moment of inertia and torsional spring constant were to be specified and the two endurance tests performed with power absorbers having characteristics at the maximum and minimum limits.

In recognition of the frequent power transients associated with helicopter operation, the specification contained Low Cycle Fatigue (LCF) design and test requirements. The test was approximately 400 hours in length and consisted of 3,500 rapid power transients. Approximately three minutes of steady-state running were required after each transient.

Sikorsky UH-60A Black Hawk in Operation in a Desert Environment

One of the most common causes for early removal of helicopter engines is performance loss due to sand erosion or Foreign Object Damage (FOD). In recognition of this problem, an integral inlet particle separator was required. The qualification requirement consisted of a 50-hour sand ingestion test, during which 80 lb (36.2 kg) of sand were ingested. Hourly transients were required to check for loss of stall margin. Maximum allowable power and SFC losses at test completion were specified.

New fuel system durability tests of increased severity were also required in the UTTAS engine specification, including a 300-hour test with highly contaminated fuel. Also required was a 300-hour fuel boost pump cavitation test at maximum speed and flow conditions.

Because of the combat environment envisioned for the UTTAS, emphasis was placed on vulnerability and survivability criteria. These included a nonvisible smoke limit and test, projectile damage design criteria, and loss of oil requirement and test. A suction fuel system, which dictates below-ambient fuel pressure at the engine inlet, was also incorporated in the UTTAS engine requirements. The absence of pressurized fuel in the aircraft lines greatly reduces the potential for serious post-crash or post-ballistic-impact fire.

The importance of keeping the engine operationally ready was also acknowledged by including maintainability, condition monitoring, and fault isolation requirements. These included guaranteed component replacement times, a maintainability demonstration, bearing accelerometers, a history recorder, and erosion/foreign object damage indicators.

Lessons Learned and Future Program Considerations

This section summarizes lessons learned from development, qualification and fielding of the T700-GE-700 engine. Regarding performance, the UH-60A has met its objectives and the T700 has been a major contributor. Particularly noteworthy is the improved part-power fuel economy. The T700 provides approximately 30% lower specific fuel consumption than previous generation Army engines of the same power class, a significant feature with today's emphasis on fuel conservation.

Advanced performance and weight requirements must be carefully selected because they strongly influence unit cost, producibility, durability, and power growth capability. Engine weight increases (with compensating IRP increases) were incorporated during the T700 program in order to control cost and introduce life improvements. The complex designs required to achieve advanced technology tend to impact engine producibility. Emphasis must be placed on durability because hardware designed to meet challenging performance and weight requirements is prone to rapid degradation in the demanding military operational environment. The following examples illustrate this point. High work compressor stages typically require airfoils with sharp leading edges, which are vulnerable to erosion or foreign object peening with resultant performance loss.

Compressor design should provide adequate stall margin after decrements for production hardware variations, field deterioration, and installation effects are applied. Close clearances increase the potential for performance loss due to rubs and for airfoil fatigue due to rub-induced excitations. Close clearances, plus the super-critical rotational speeds of small engines, dictate emphasis on accurate rotor balance at assembly and on design features which retain proper balance in the field. The small orifices and passages of hot section cooling schemes are vulnerable to plugging by ingested sand particles, oxidation, or upstream loss or rub coating. Design attention, as well as technology advances, are needed to reduce the magnitude of these effects in small, high performance turbine engines.

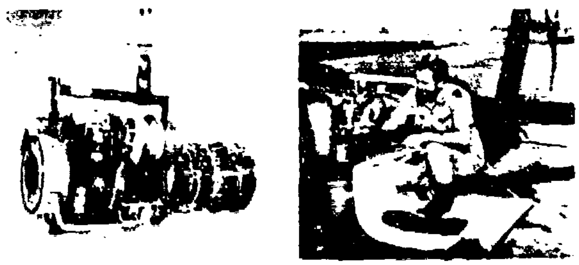
In addition to performance/weight vs. cost, producibility, and field durability tradeoffs, there is also an important performance/weight vs. power-growth-potential tradeoff which should be considered in establishing requirements. High turbine temperature associated with advanced performance goals increases the difficulty of subsequent temperature increases for growth. The integral inlet separator adds to the growth challenge by inhibiting airflow increase through addition of compressor stages. The first step of T700 power growth is being met with the Navy T700-GE-401 and Army T700-GE-701 engines, primarily by increased turbine temperature. Further growth will require technology advances in such areas as power-turbine-driven booster staging and high temperature durability of small turbine airfoils.

The T700 control system was successful relative to the goal of relieving pilot workload. However, the aircraft test program revealed that constant speed governing contributed to helicopter rotor speed droop following large collective pulls with the rotor decoupled. The collective pull causes power turbine speed to increase slightly prior to main rotor reengagement. As a result, fuel flow is reduced by the governor, thus amplifying the droop tendency following reengagement. Further advances in control system technology are needed to provide improvements in this area. In the future, engine testing which simulates autorotation and subsequent reengagement is needed to evaluate the effects of these conditions early in the development program. The power turbine overspeed control system successfully prevented excessive overspeed in a qualification test designed to verify this capability. Subsequent analysis indicated that at high altitudes peak speed reached following load loss could exceed minimum burst speed. Design changes being incorporated in the T700-GE-701 will ensure overspeed protection at extremes of the operating envelope as well as at normally encountered altitudes.

The durability testing was quite successful in finding and fixing problems before production. The endurance, LCF, and sand erosion tests were all beneficial for this purpose. However, LCF testing was not initiated until the final year of the development program. Prior endurance tests were to the 150-hour test cycle, which is mostly at steady-state conditions. Earlier cyclic testing would have further increased the number of LCF fixes incorporated into the first production engine. Also, subsequent experience has shown that the bearing and power takeoff components of advanced small engines require additional evaluation beyond that provided by engine endurance testing. Lube system simulator plus instrumented engine testing should be conducted early in the development program to define the operating environment of these components. Fatigue testing of critical elements should be performed to establish life.

There are several tradeoffs associated with making the Inlet Particle Separator (IPS) an integral part of the engine, in addition to the growth constraint mentioned above. The integral approach has the major advantage of development and qualification of the IPS with the hardware it is designed to protect. However, the integral IPS increases the engine anti-icing air requirements, with a resultant power penalty in icing conditions. It imposes constraints on aircraft inlet design by requiring interface with IPS features and may prevent inlet optimization for multi-engine installations. It cannot be removed when extended operation from improved airfields is envisioned. Hence the inherent weight and inlet pressure loss penalties associated with any inlet protection system are always present with an integral design. The scavenge passage of the engine-integral approach can be a path for reingestion of debris or exhaust gas into the compressor inlet. In view of these tradeoffs, the integral IPS question should be carefully considered for each new program. It is still considered the correct approach for U.S. Army engines. However emphasis in future programs should be placed on minimizing the adverse tradeoffs cited above.

The suction fuel system capability has been achieved by the T700/UH-60A system, with resulting major safety benefits. However, the aircraft development effort revealed the need for the following special design and test considerations. During aircraft operation with hot JP-4 fuel at high altitude, air is released in the aircraft fuel lines, especially at locations where large pressure drops occur. This air can accumulate in horizontal lines and may be released as a large bubble into the engine fuel boost pump, resulting in flameout. Suction fuel delivery systems should be designed to minimize these effects. Shutdown in hot ambient conditions, plus engine heat soakback into the fuel, can also cause vapor buildup in a suction system, with resultant starting problems. Fuel system leaks are a source of air entrainment with a suction system. Leakage checks should place the fuel system under a vacuum to simulate actual conditions. Because of the strong installation influence on suction fuel system performance, the aircraft/engine interface conditions should be defined as completely as possible. Specifically, the engine program should include a suction fuel system for the altitude performance testing to provide engine test verification of the pump's capability under high V/L conditions. The aircraft program should include



Two T700-GE-700 Turboshaft Engines of 1,560 Shaft Horsepower are Installed in the UH-60A and AH-64

verification that fuel temperature and vapor content at the engine inlet are no higher than the engine requirements throughout the operating envelope.

Early data following fielding of the UH-60A indicate that the T700 maintainability features will provide a significant improvement in operational readiness. The engine is being maintained as an on-condition system without a scheduled overhaul interval. Required inspections are limited to a 10-hour check of condition-monitoring features and a 500-hour "on-wing" borescope inspection. Many of the fault indication ideas evaluated in development proved not to be feasible for production due to limited accuracy, questionable reliability and adverse impact on engine cost and weight. The history recorder on the production engine has counters to record engine hours, power transients, and time/temperature effect. The modular design of the T700 plus the availability of these life usage indicators, dictates improved record-keeping in the field to ensure that life usage data on removed modules, as well as engines, are maintained. If this difficulty can be overcome, the history recorder will provide a much more accurate means for monitoring hardware life compared to the traditional approach based only on engine operating hours.

Conclusion

The T700 engine was developed and qualified to meet several new U.S. Army requirements. The resultant configuration is already providing far-reaching benefits. Lessons learned from the program will be applied in the definition of future engine specifications, as part of the continuing process of optimizing requirements for the unique military helicopter environment.

SECTION 2

ENGINE DEVELOPMENT PROGRAMS

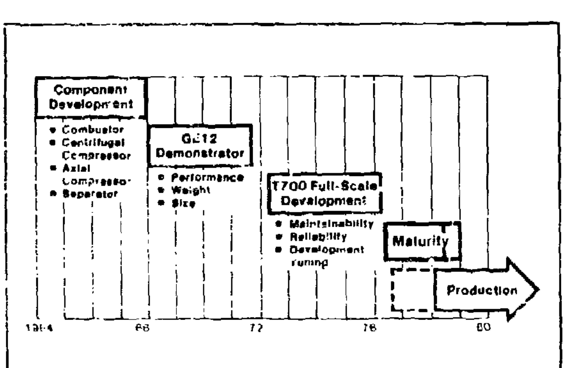
by William J. Crawford, III

This section discusses the T700 engine development program generated in response to the U.S. Army requirements as described in the previous section of this paper. It is presented from the engine contractor's point of view.

Of all the many influences involved in the success of a development program, there are three factors of major importance from the engine contractor's viewpoint:

- He must have an established Technology Base from which to draw for program development and engine design
- He must have a thorough understanding of the customer's needs
- He must have the benefit of "Value Added Management" depth to meet the challenges requiring innovation and invention.

From the beginning of the UTTAS engine effort, the process followed by Army program management was directed to reducing the risk of applying state-of-the-art technology. Advance verification of the performance of small engine components was critical to the decision to proceed to full-scale development since the higher pressures and temperatures required had not previously been achieved in this size engine. Evolution of the engine progressed through several years of component development followed shortly thereafter by a competitive 4-year engine demonstrator program that lead to the full-scale development program.



Engine Evolution

General Electric approached the performance challenges presented by the small engine design by adapting technologies already developed and proven on its large commercial and military engines. These adaptations were combined with ideas developed through Army-sponsored small engine research and were subsequently component-tested in the initial stages of the program.

Once these component technologies were successfully run in the demonstrator engine program, GE set out to define customer needs for its own perspective in order to determine total engine design that would be responsive to the user's operational needs. Consultations were held with Army personnel at every level and discipline to evolve the overall engine

design features, later to be proposed as the T700 design. Additional perspective in maintainability was obtained by conducting an Army-critiqued "maintenance demonstration" on the performance-oriented demonstrator engine. This gave the design areas hands-on experience with actual hardware.

Applying innovative approaches to the user's "lessons learned" problems was the key to achieving the maintainability, reliability and other "ilities" characteristics unique to today's operational T700. The inlet separator, self-contained lube and electrical systems, and spring-clip line clamps are examples of this input/contractor innovation process.

Development Program

The specificity of the U.S. Army requirements in the various disciplines - i.e., maintainability, reliability, vulnerability, etc, - demanded that all the disciplines input to the engine design process from the beginning. Representation from the separate specialty areas actively participated in the generation of the design and were responsible to see that their particular requirements and interests were present in the final design. Drawing release for manufacture required review and sign-off by all areas. This was unprecedented, as was the veto power that each had, thus assuring that their discipline interests were protected. Conflicts were arbitrated by the Design Review Board with final decisions by Engine Program Management.

Continued involvement of all the disciplines was also reinforced by the reporting and demonstration requirements set down by the Army. For example, three maintainability demonstrations were required. The first two were conducted by GE personnel monitored and measured by the Army. One was scheduled early in the program and the other at the midpoint. The final demonstration was an evaluation performed by U.S. Army personnel who were representative of training and skill levels of field organizations. The engine used was one of the MQT engines following its post-test inspection. The results verified that overall required objectives were met, and also indicated areas where improvements could be made. In a similar way, factory engine reliability was tracked and reported monthly against a goal requirement at MQT. Contract incentive payments were contingent upon attaining these goals/requirements.

The development test program incorporated all the specialized environmental, overspeed and overtemperature tests standard to the industry. The Army requirements also specified other engine tests that were tailored to the helicopter application such as sand ingestion, power turbine loss of load/overspeed protection and a low cycle fatigue test. Going beyond the initial program requirements, an additional series of 25-hour engine/aircraft integration tests was included later in the program. These tests subjected the engine and the aircraft mounting, inlet and exhaust components to abusive vibration stresses concurrent with operation to the MQT cycle. The development program required that a total engine factory test of at least 2,500 hours be attained by the Preliminary Flight Rating Test, and a minimum of 7,500 hours at MQT. The latter was exceeded by approximately 1,500 hours. The development test program was culminated with a double MQT, qualifying the engine with two different fuels and oils, as well as providing two samples for durability evaluation.

Maturity Program

Overall UTTAS Program timing planned by the Army provided for Competitive Test (GCT) of the two different aircraft each of which powered by the same configuration T700. The test was begun at about the same time that engine MQT was completed. Under prior program standards, engine qualification would be considered complete at this time and the engine would have been committed to production. At this point, a post MQT program was initiated with the goal of accumulating additional endurance experience and subjecting the engine to more LCF testing.

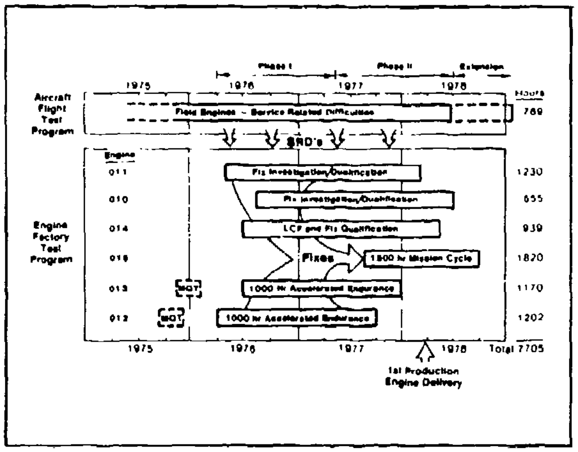
The overriding purpose of this Maturity Program was to provide a mature, reliable engine prior to full rate production. To accomplish this, the following objectives were established:

- Develop high initial Mean Time Between Failure-Require Overhaul (MTBFRO)
- Establish sound field maintenance procedures and intervals
- Identify unique installation-related failure modes
- Establish programs for smooth transition to production manufacture

The approach selected was to conduct accelerated, severe, abusive tests so that the required production target dates were assured.

A secondary benefit of the Maturity Program was that it provided a highly valuable period to resolve residual problems uncovered in the field and factory programs and any that might evolve from the aircraft GCT. In addition, a smooth transition to Production through Producibility and Manufacturing Technology programs was made possible, as well as the implementation of cost reduction programs prior to production.

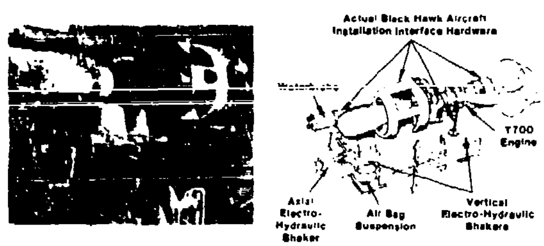
The Maturity Program was divided into two major parts for maximum effectiveness.



Maturity Factory Program

- Factory
  - Two 1,000-hour accelerated endurance tests
  - 1,500-hour mission test cycle
  - Low Cycle Fatigue test (3,500 cycles)
  - Service Revealed Difficulty/ Fix qualification program
  - Qualification of Production Parts List
- Field
  - Support accelerated aircraft test program
  - Update prototype engines to latest configuration
  - Repair/overhaul as necessary
  - Perform high time analytical inspections

Immediately following the completion of the Qualification Program and the maintainability demonstration, the two MQT engines were reassembled and began running to accumulate the 1,000 hours on each. Any "failures" during this period of intensive endurance testing were considered as successes since they identified weaknesses in the system that could be resolved, endurance tested, and qualified for initial production. Approximately one year after initiation of these 1,000-hour endurance tests an Accelerated Simulated Mission Endurance Test (ASMET) was initiated.



Simulates Engine/Aircraft Installation Operating Conditions

1,500-Hour Mission Cycle Test

At this point it might be best to highlight the 1,500-hour ASMET mission cycle test vehicle. As indicated in the illustration the engine was supported by the actual aircraft mount system and also had the airframe inlet system, exhaust system and control/sensing and fuel connections applied to simulate the aircraft installation. The entire test vehicle was installed on a vibration test stand and simultaneously subjected to the full range of frequencies characteristic of the helicopter installation and an engine operating cycle of increased severity containing significantly more transient operation than the previously conducted endurance tests.

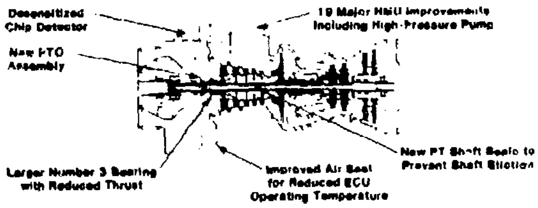
Concurrent with these factory tests, flight test engines were generating service problems (SRDs). Subsequent design improvements made against these SRDs were incorporated into the ASMET engine to make it as close as possible to the production configuration.

All field operations support during the Maturity Program was provided by the Integrated Logistics Support Management Team. This effort included deployment, at a number of operating sites, of necessary spare parts, special ground support equipment, and technically qualified field representatives. Engines used during this phase of the field program were Flight Rated engines updated to MQT configuration plus some maturity design improvements. The update of these engines and subsequent initial repair engines were processed at the General Electric Depot Service Facility.

Coordination of field maintenance tasks for installed engines was conducted with the airframer. Technical Manuals and Repair Parts and Special Tools Listing were reissued reflecting updates to the mature engine configuration immediately after MQT. All field maintenance, troubleshooting and repair was conducted against these documents to "test" them for accuracy and adequacy in the hands of the user. Depot manuals reflected the same updates, again as a base for improvements determined from experience.

Maturity Program Results

A number of field problems were exposed during the severe competitive evaluation of the air vehicles. The most important of these corrected were redesign of the Number 3 Bearing and the Power Takeoff Assembly. Other improvements included the Hydromechanical Unit (fuel control) and a new high pressure fuel pump, lower sensitivity chip detector, new power turbine shaft seals, and an improved airseal to reduce Electrical Control Unit operating temperatures. All of these changes were developed and qualified for first



Production Initiated with all GCT In-flight Shutdown and Flight Abort Causes Eliminated

Durability and Reliability Improvements from GCT

Rate per 1,000 Hours

Description	Recorded UH-60A GCT	Current Field Status
Oil Samples (SOAP)	81.88	0
Oil Additions	52.55	16
Oil Filter Replacement	8.75	0.13
False Chip Lights	1.25	0.13
LRU Malfunctions	11.25	1.90
Total	155.63	18.16

88% Reduction in Maintenance Actions

Reduced Maintenance Actions

	Endurance Hours	Fixed – First "T" Engine
• 1000 Hours – Engine 012		
– Number 3 Bearing	C8	✓
– Stage 1 Bucket Trailing Edge	300	✓
– Stage 2 Bucket	488	✓
– Stage 2 Turbine Air Seal	455	✓
– PTO	621	✓
– Stage 3 Turbine Air Seal	888	✓
• 1000 Hours – Engine 013		
– P_3 Bellows (HMU)	1144	
• LCF Engine 014		
– Power Turbine Casing Cracks	360	✓
• 1800 Hour ASMET Engine 015		
– ECU Bracket	60	✓
– P_3 Hose	60	✓
– Swirl Frame/History Recorder Mount	378	
– [illegible] Housing	590	
– Ignition Exciter Bracket	968	
– [illegible] Valve Fuel Manifold	1080	

Costly Field Retrofit Programs Avoided

New Failure Modes Identified and Fixed

production engine introduction, thereby eliminating all known in-flight shutdown and mission abort causes experienced in GCT.

First hand observation of field operations also had indicated areas where maintenance procedures and some design features could be modified - or in some cases be eliminated - to decrease field maintenance actions by as much as 85%. It is interesting to note at this point that the top three problems on the Army's list were the frequency of adding oil, oil sampling and oil filter changes. All items the engineer had little concern about previously!

The accelerated factory tests did identify new failure modes as indicated by the chart. Most of these were fixed for first engine production. Without the maturity experience these problems would have remained hidden for one to three years or more based on average military use. The result would have been expensive programs of a fail-analyze-fix nature followed by retrofit.

Maturity program testing also led to a number of design changes for performance improvements. Following changes to the compressor and combustor, throttle stall margin and deceleration stall margin were improved by more than 30%, greatly enhancing operational suitability. The power turbine was improved to increase its efficiency as well as for better maintainability and reduced cost.

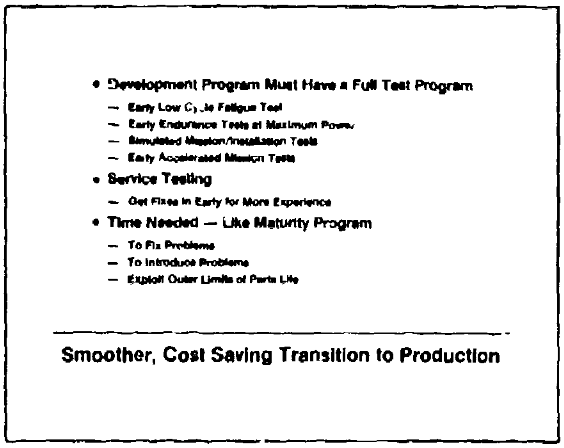
Logistics support benefits derived from the results of the Maturity Program were many. Materiel support requirements for the fully operational system were established for spare parts and spare engines, and ground support equipment was designed and qualified. Support facilities were set up and qualified as required, such as the depot/overhaul shop and central warehousing. Field procedures and depot repair procedures were developed and verified. Documentation was updated to reflect actual field operation and issued before first fielding the production aircraft. In effect, an experienced organization and system were in place to support the UH-60A from the very beginning.

Lessons Learned and Future Considerations

As noted in Section 1, the test program generated for the T700 Development Effort did have some of the special tests necessary to expose short term LCF, thermal cycle problems, and stress rupture. However, not only was this testing conducted too late to be of much benefit to the development effort, but there was not enough of it to provide a realistic picture for verifying the required design life.

The Maturity Program was necessary to exploit this kind of test and serves as an excellent benchmark from which to create future development programs. Endurance testing through MQT cycle tests finds durability problems, but they take a long time to surface and many are stress-rupture oriented. The ASMET was a sound compliment to the Development Program. It provided valuable exposure of the engine/airframe interface hardware and did uncover a number of problems, but fewer than anticipated. The test cycle used in this ASMET was tailored to the Black Hawk mission giving about 3:1 severity compared to field operation.

Based on the experience gained in the eight years of test, a new accelerated test cycle has been designed which provides a better balance of LCF and maximum temperature operation. The Accelerated Mission Test (AMT) has shown tremendous payoffs. It is



Lessons Learned

compared with the 6-hour MQT Cycle and LCF and ASMET Cycles in the chart. Older test efforts spent too much time on tests which were not representative of user operation. A significant part of current GE test programs for all engines continues to be the AMT.

Future test program planning should begin testing earlier. Strong considerations should be given to including the AMT as part of the Development Qualification Program.

Conclusion

One of the best measures of the success of a program is the level of Engineering Change Proposal (ECP) activity in the first few years of service. The T700 has now been in production for three years and has accumulated more than 65,000 operating hours. The engine stands out for the low ECP activity. Only 12 ECPs have been approved in the three years. And, only one of them is a new "failure mode" identified and fixed. One was for Manufacturing Producibility. The other 10 were residual from the late stages of the maturity effort - 3 uncovered by ASMET and 7 from the competitive flight test program. The first production model of the T58 engine (same approximate size and horsepower) of twenty years ago had 66 ECPs for the corresponding period.

The excellent detail program plans defined by the Army for the UTTAS engine made winning the competition for development a tough challenge. Once embarked on the program, however, valuable experience was gained that resulted in successful development of an operationally ready engine for the user.

Key elements of this success were:

- Full development under what should be called a normal specification
- Extensive flight testing - with the real engine and installation - provided identification of most of the field problems.
- Time to develop fix-plans and a comprehensive manufacturing plan.

DISCUSSION

W.Heilmann, Ge

Could you describe some of your reasons for low cycle fatigue testing including high speed and consideration of powder metallurgy discs?

Author's Reply

High speed tests uncovered a problem area that was not anticipated in the design analysis, clearly justifying its value. In addition, of the failure test made with the powder metallurgy discs, no failures were encountered involving defects.

FUTURE TECHNOLOGY AND REQUIREMENTS FOR HELICOPTER ENGINES

by

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SUMMARY

Engine design features and the technology needed to produce them arise from the operational requirements of the user services. The paper discusses the design considerations and technology needed to meet possible future operational requirements and describes the relevant technological work being carried out or supported by the UK Ministry of Defence (Procurement Executive). Also considered are the trade-offs which become necessary in seeking a suitable solution to conflicting needs. Finally the sizes of engines required to meet likely aircraft applications are examined.

1. INTRODUCTION

The introduction of the gas turbine caused a revolution in helicopter range/payload capability due to the dramatic improvement in engine power to weight ratio, which increased by a factor of about 3:1. Since that time helicopters and their component parts have steadily improved, and should continue to do so for the foreseeable future. As the general capability of helicopters has increased so too have the expectations of the users. One of the aims of the Ministry of Defence (MOD(PE)) is to ensure that the technology necessary to fulfil the expectations of the UK Services is available when required. This is essential if gestation periods for new engines are to be compatible with those for helicopters. To achieve this aim MOD(PE) and Industry work in collaboration on a broad spectrum of research and technology programmes. In the gas turbine field several such programmes have been, or are, underway and some have been directed at helicopter engines in particular. Inevitably some helicopter requirements cause conflicts in engine design and trade-offs must be made. It is intended to indicate typical requirements for both military and civil helicopters and to show the types of trade-off which may be made to produce a successful, general purpose gas turbine. Important influences on these trade-offs are the rating structure of the engine and the certification tests required to clear these ratings. Care must be taken that designing to meet the operational requirement and designing to meet the certification rules do not lead to different answers.

2. PREVIOUS PREDICTIONS

It is instructive to go back to records of previous AGARD conferences on helicopter propulsion, held in 1968 and 1971, and look at predictions of the improved technology of engines that would be in production in 1980 (Ref 1). Based on cycle and component analyses, it was forecast that, at the design point, a Specific Fuel Consumption (SFC) of 85μg/J (0.5 lb/SHP hour) an overall pressure ratio of about 14 and turbine inlet temperatures of 1300-1550K were to be expected. This compared with values then current of about 100μg/J (0.6 lb/SHP hour), 12, and 1200-1450K. We can record with satisfaction that these predictions have in the event proved correct. Figure 1 illustrates the variation, at the design point, of specific power and specific fuel consumption with cycle pressure ratio and temperature. The precise shape of these curves depends upon the assumed cycle and component efficiencies. The choice of these is in turn, governed by the need to maintain those good handling qualities without which any engine, however efficient at its design point, would be little more than useless.

A development not predicted in the AGARD papers of 10 years ago was the advent of torquemeters of a type suitable for engine condition monitoring and deterioration checking. Also unforeseen was the full potential of micro-processors, then in their infancy. The implications of these devices are dealt with later in this paper. It could be argued that these further developments, which were not discussed ten years ago, have been as significant as those which were given great emphasis. We shall be interested to see whether any predictions made at this conference manage to strike the right balance. In ten year's time, with the benefit of hindsight, we shall know.

3. GENERAL DESIGN CONSIDERATIONS

3.1 Range and Endurance. The primary requirement for any aircraft is that it can carry its load over a prescribed distance or for a given period of time.

To provide a given helicopter with a good operating range and endurance the engines must have a low fuel consumption, particularly at cruise conditions. Two principal factors decide the level of fuel consumption. Firstly there is the efficiency of the engine itself, as determined by its cycle and component efficiencies. Secondly there is the matching of the engine to the aircraft, particularly the sizing of the engine.

The design and technology requirements for high cycle efficiencies, in terms of component performance, and higher pressure and temperature ratios are well enough known. Technology programmes are now in hand in both these areas and para 5 describes some of the work being carried out to develop cooled turbines suitable for small engines. Whilst these developments in small military engines are very important, attention must be paid to their effect on initial and life-cycle costs, and to ensuring that gains in design point cycle efficiency are not made at the expense of engine handling qualities, at off-design conditions.

The question of engine sizing is one of considerable complexity and some controversy. Two schools of thought seem to exist, and for any given application the right compromise must be struck. If low specific fuel consumption is an over-riding requirement then it is necessary to operate at the lowest practical point on the SFC curve, ie at as high a proportion of maximum power as possible. This means using a small engine running at comparatively high power levels. Obviously there is a penalty to be paid in engine life, when this is compared with the use of larger, less highly rated engines as proposed by the second school of thought. Such larger engines would be able to cope with engine failures on multi-engine helicopters without the need for contingency ratings, but would carry penalties in terms of weight and SFC, and possibly aircraft range. The relationship between optimum engine sizing, the rating structure and qualification requirements is discussed further in para 6.2.

3.2 Agility and Engine Handling. Aircraft agility and problem-free engine handling are necessary for all helicopters, but especially for those on the battlefield or landing on moving ships. From the engine aspect, agility principally requires sufficient installed power to allow rapid aircraft climb and acceleration, a question again of engine sizing. Also important is a rapid response of the engine to a demand for power. Engine response is of particular concern in cases of engine failure on a multi-engined helicopter, when the remaining engine(s) must reach Contingency or Maximum power before rotor speed has fallen dangerously. Although there is no reason to suppose that the response times of the most recent engines are too slow for present day needs, the introduction of light weight helicopter rotors, which would have less stored energy, may bring about a requirement for faster engine response times. The prime technical requirement for good engine handling and response times is an adequate compressor surge margin. Engine response would also benefit from improvements in the mechanical design of engine rotating components, giving reduced moments of inertia and a greater angular acceleration for a given level of over-fuelling.

An agile helicopter must not only accelerate rapidly but also decelerate rapidly without gaining height; low flight-idle power levels may therefore be desirable. It must also be ensured at the design stage that the oil system will function for adequate periods of operation at low or negative "g".

3.3 Availability. Two major factors affect this property. They are maintainability and reliability.

3.3.1 Maintainability. Ease of maintenance is important both in the battlefield and on board ship. Further moves towards "on-condition" maintenance are envisaged, using a range of engine health monitoring techniques. For first and second line maintenance however, the main requirement is to be able to remove and replace engines and modules in minimum time, using only the smallest and most basic tool kit. It is doubtful whether new technology is needed to achieve these goals; however, it is vitally important to give careful attention to them at a sufficiently early stage in the design of the engine. Maintainability must be designed in, and cannot be added later as a "bolt-on-extra".

Further moves towards on-condition maintenance, as opposed to the use of fixed life components, will depend on improved techniques of engine condition monitoring, particularly in the fields of low cycle fatigue, thermal fatigue and creep usage monitoring, and non-destructive testing. Improvements in the bore-scoping of gas generator turbines are also highly desirable, difficult though this is in engines with reverse flow combustors.

Whilst modular construction is now the accepted practice for aero engines, it is desirable that engines rebuilt from modules of differing lives should not require running on a test bed prior to installation, and for multi-engined installations it is desirable to be able to torque match the engines without the need for a flight test.

3.3.2 Reliability. Reliability may either be considered in terms of the helicopter or of the engine alone. The reliability of the helicopter is determined partly by fundamental choices such as the number of engines. A multi-engined helicopter will have a better chance of completing or at least returning safely from a mission than a single-engined one. The number of defects per operating hour will however be higher, assuming the reliability of each engine installation to be the same.

The reliability of an individual engine depends on many factors: not only the most obvious ones of quality of design and manufacture, but also the rating of the engine and the lifing and maintenance policies adopted. The basic quality of design and manufacture, fitness for purpose, is validated by the programme of qualification testing, all of which may be considered as part of the reliability activity which is now accorded special attention in development programmes. The rating of the engine, resulting from the trade-off established between life and power and the mission profile, will determine the potential life between reconditioning or replacement of the engine and its modules. As parts approach the end of their potential lives, the probability of failure increases. The lifing and maintenance policy will determine how far engines are allowed to venture into this region before being rejected as time-expired.

Reliability is therefore not a characteristic which can be developed independently of other engine parameters. It is one of the many interacting variables which influence the design of the engine and its cost of ownership.

3.4 Life Cycle Costs. The Services require helicopters which not only meet their technical requirements, but do so at minimum cost. In the past, emphasis has been placed on development and initial production costs with estimates being based on data from previous projects. More recently,

attention has been given to Life-Cycle Costs (LCC), which also take account of the operation and support costs.

The extent to which the aim of low life-cycle cost may be achieved depends on the priorities given to the other requirements, on how intensively the aircraft are used, and on the total number of engines produced, for both military and civil customers. A simple engine may be cheap to buy, and so may its spare parts. However, a more sophisticated fuel-efficient engine used intensively might save the difference between acquisition costs, through savings in fuel. The engine rating also affects life-cycle costs; the benefits to cruise SFC, and hence operating costs, of a highly rated engine may be offset by higher support costs.

During the early stages of a helicopter project, many complex and interacting variables, all of which influence life-cycle costs, have to be analysed. The life-cycle costs for the various helicopter designs considered during these early stages can in principle be estimated using mathematical models. Although there are many such models, there is considerable difficulty in obtaining reliable cost data of sufficient accuracy to demonstrate the sensitivity of LCC to changes in engine design. It follows that small changes in predicted costs need to be treated with caution.

Although life-cycle cost estimates are carried out separately for the engine, they must not be considered in isolation. The aim of the operator should be to reduce the life-cycle costs of his helicopter fleet. This may not always be achieved by using the engine with lowest life-cycle cost. The use of an advanced engine with higher life-cycle costs but with lower weight and fuel consumption might so influence the design of the aircraft that it permits a substantial reduction in the life-cycle cost of the complete system.

The following might be a list of life-cycle cost priorities for future engine design:

a. Identify the most sensitive engine-related cost factors, for example SFC or power/weight ratio, and design the engine installation including intake, transmission and exhaust to minimise helicopter LCC, whilst using an engine that will be close to the technological state-of-the-art at introduction into service. (Ref 2).

b. Design for minimum total component manufacturing costs, thus also reducing spares costs (Ref 3).

c. Design to allow repair/replacement of defective parts as close to the front-line as possible (Ref 4). Modularity is an example of this concept, but care must be taken to ensure that hardware benefits are not cancelled by increases in paper-work and data recording.

d. Develop a logistics model, based upon component failure information/predictions, that will allow an optimum servicing policy and organisation to be developed. This would take advantage of Item c above and also make the best use of the equipment that had evolved and actually existed.

e. Use "on-condition" monitoring techniques, planned in such a way as to minimise shop visits and LCC. The model of Item d should be updated regularly to reflect current service experience. This model should use only the most significant lifed components for data.

3.5 Flexible Use of Ratings. At present UK engines are certified with the power levels above Maximum Continuous available for only a limited time per flight, a flight being defined as from take-off to touch down. To give the maximum flexibility of operation it would be desirable to dispense with these somewhat artificial time limits. They are imposed, not because any immediate dreadful consequences are certain to occur if they are ever exceeded, but so that the engine is unlikely to fail from hot end distress in its authorised time between overhauls. The use of on-board engine usage monitoring systems would enable greater flexibility to be granted with safety. Some means of discipline in engine usage would still be necessary to ensure that an adequate life was achieved in practice, but the rigid manner in which the limitations are currently expressed in the UK could be relaxed. The relationship between rating structure and qualification requirements is discussed further in para 6.2.

3.6 Extended Fuel Capability. The current range of gas turbines is able to operate satisfactorily for extended periods on all currently available gas turbine fuels ranging from high flash point (F-44, AVCAT, JP-5) to wide cut (F-40, AVTAG, JP-4). Many can also run for limited periods on diesel and/or gasoline fuels. Future engines will also need to accept gas turbine fuels which, as a result of worldwide difficulties in supply and local variations in fuel properties, may not meet current quality standards. Inevitably the ability to run on a wide range of fuels might necessitate some compromise in design which would not be necessary if the engine were required to run on high quality turbine fuel only.

Crude oils vary both in the nature and proportion of their constituents. As the availability of crude oils changes both the physical and chemical properties of future fuels will change. Figure 2 shows distillation curves for typical fuels currently available (Ref 5). The extent of the distillation (temperature) range of any fuel is representative of the availability of that fuel from the parent crude oil. It can be seen from Figure 2 that fuels such as AVTAG, MOGAS and DIESO have a wider distillation range than AVTUR or AVCAT. A hypothetical future wide cut fuel is also indicated. Some properties which may change are given below:

Freezing point: raising of the freezing point could lead to fuel flow problems in cold conditions both in flight and on the ground, possibly necessitating tank heaters.

Specific energy: reductions in specific energy could negate improvements in engine SFC e.g. a general reduction from 43 to 42 MJ/kg which is only within the range of current fuel specs, would lead to a corresponding increase in SFC, of 2.4%.

Aromatics: increases in aromatic content would affect the performance of elastomeric seals, and would also lead to increases in the amount of heat radiated to combustion chamber walls due to increased flame luminosity.

Viscosity: higher viscosities could lead to poorer atomization and vaporization and thus to starting difficulties, and "streaky" combustion.

Other obvious important properties which could change for the worse are vapour pressure, thermal stability and lubricity.

4. ENVIRONMENTAL HAZARDS

It has been said that a helicopter engine has to cope with two environments; the one that was there before it arrived, and the one that it brings with it. An examination of typical helicopter environmental factors tends to produce a list similar to that in Table 1. The three Services, Army, Air Force and Navy generally require the same features but with a different balance of priorities. The operating environments tend to fall into two broad categories. The first is the maritime environment in which Naval and some Air Force aircraft operate, and the second is the overland/battlefield environment in which Army and Air Force aircraft principally operate. It may also be argued that although some of these hazards exist for civil helicopters the severity may not be so great.

4.1 Intake Protection. The dirty atmosphere produced by main rotor downwash during hover near the ground, and the sensitivity of small gas turbines to impact and abrasion by ingested foreign matter, make the provision of engine intake protection highly desirable. Stoneguards and sand filters are available for the current generation of engines as optional 'role' equipment. Future engines will require a better dust particle separation efficiency in order to avoid the blockage of cooling holes in film-cooled turbines, as well as further to reduce impact and abrasion damage in compressors. A technology programme, supported by MOD(PE), has examined a number of different configurations of separator.

An intake separator should of course also provide protection against impact from birds, ice shed from the airframe, and hailstones. Provided that ingested objects are reliably directed away from the turbo machinery, the blading may be better optimised for performance. Further advantages that may accrue from a correctly designed separator are an improvement in intake distortion due to better conditioning of the intake air and, in the case of Naval aircraft, less internal corrosion caused by salt water.

4.2 Icing and Salt Corrosion. Resistance to icing and salt corrosion are particularly important for maritime helicopters. A number of helicopters operated by the UK Services now have limited clearance for operation in icing conditions. The latest UK helicopter engine, the Rolls-Royce Gem, is fully cleared for operation in icing conditions; anti-icing air is used to heat the intake, and ice ingestion tests have been carried out satisfactorily. Extension of the clearance to fly in icing conditions is therefore dependent on advances in airframe and rotor rather than engine technology.

Corrosion as a result of salt water ingestion is a familiar problem in engines operating over or near the sea. Corrosion is best avoided by a suitable choice of materials and protective treatments when the engine is first designed. A strong incentive to do this is the knowledge that the engine will be submitted to a stringent qualification test. At present the US salt corrosion test is considerably more severe than the UK test. A Rolls-Royce Gem engine, incorporating a number of modifications has recently completed a test to the US schedule with very encouraging results. Consideration is now being given to adopting the US requirement for UK qualification of engines to be used in marine environments.

4.3 Noise. A low noise output is helpful not only to commercial operators but also to enable a battlefield helicopter to avoid detection. Although the major and most characteristic element of helicopter noise is produced by the rotor, a reduction in engine noise is nonetheless desirable. Intake separators and exhaust IR suppressors will in themselves give some degree of alleviation. The design of the engine installation is also an important factor. Any further measures, such as acoustic treatments, need to be carefully evaluated as they could carry a considerable weight penalty.

4.4 Battle Damage Survivability. There is little that the engine designer can do to reduce the vulnerability to battle damage of the internal parts of the engine, apart from taking such measures as strengthening the outer casings. Careful attention should however be given to the routing of pipes, cables and controls to make them less vulnerable. External mechanisms, such as variable compressor geometry are also weak points. If possible the design of the engine installation should be such as to make it unlikely that more than one engine will be struck by a single projectile. Local armour protection is possible, but d · account should also be taken of weight.

4.5 Suppression of Infra-Red Radiation. An important requirement for battlefield helicopters is protection from heat seeking missiles. Heat is emitted both from the hot parts of the engine and from the exhaust gases. Methods must be found to minimise both, without unduly compromising other requirements. Fundamentally, the quantity of heat rejected, in the exhaust, at a given engine power can be reduced by the use of more efficient engines. A further reduction in

radiation from the exhaust can be achieved by lowering its temperature, for example by diluting it with cold air in an exhaust suppressor. Possibly, air from the intake separator could be used for this purpose. Heat emission from hot metal parts may be reduced by conventional shielding, and by an exhaust configuration which avoids a direct or indirect external view of the final turbine stage.

The means of infra-red suppression which are adopted need to be considered in the context of the engine installation as a whole if unnecessary performance and weight penalties are to be avoided. It is therefore vital that the need for such devices is considered at an early stage of aircraft design.

5. TECHNOLOGY PROGRAMMES

To maintain the rate of technical advance needed to achieve more exacting requirements, a comprehensive programme of basic research is vital. It is not always practical, however, to apply research results directly during the design and development of a new engine. The technology demonstration programme is a stepping stone between basic research and full development. It provides an engine environment, more representative than rig tests, for establishing new technological features and thus reduces development programme risks. Further, a broad spectrum programme will enable engine design options to be kept open until a specific application is fully defined.

5.1 High Temperature Small Turbine Unit. Helicopter engines manufactured in the UK have until now had uncooled turbines, and consequently have operated at lower turbine entry temperatures than are usual for other military engines. The High Temperature Small Turbine Unit (HTSTU) consists of a Rolls-Royce Gem HP module modified to accept a special combustor and cooled turbine, together with an air feed system, for evaluation of various designs of cooled turbine.

The introduction of cooling to small turbines poses several difficulties. The small, low aspect ratio blades present problems in achieving sufficiently close seal clearances to maintain efficiency. The provision of cooling passages creates manufacturing problems, which must be resolved whilst still seeking to minimise production and life cycle costs. The need for thicker blades, to accept cooling passages, results in fewer blades both for aerodynamic reasons and in order not to exceed acceptable levels of disc stress. This must be achieved with minimim loss of aerodynamic efficiency. In such blades thermal fatigue rather than creep is likely to be the failure mode.

By mid-1980, the original objectives of the programme had been achieved. At the target turbine entry temperature satisfactory blade metal temperatures had been achieved at progressively increasing turbine efficiencies. Further objectives for the HTSTU were set in April 1979, which included further increases in efficiency and life of the overall system, and the evaluation of representative engine components. Greater efficiency is being sought from further increases in blade metal temperatures, improved sealing, acquisition of data on blade tip clearances and a reduction in parasitic losses. Features developed in the HTSTU are to be endurance tested in a core engine for life evaluation. This will be a cyclic test, with representative acceleration and deceleration rates. Most of this programme is due for completion by mid-1981, but the endurance testing will continue until 1982.

5.2 Digital Electronic Control Systems

In 1979 a contract was placed by MOD(PE) with Rolls-Royce Ltd, to develop a demonstrator digital control system for helicopter engines, using the Rolls-Royce Gem as the demonstration vehicle. The suppliers of the equipment are Dowty-Smiths Industrial Controls (DSIC), a joint company in which Dowty and Smiths Industries are partners. The control system, which is micro-processor based, is expected to give significant savings in cost of ownership when compared with hydromechanical or analogue electronic systems. There should also be much less routine maintenance and calibration. Improved torque matching will reduce the pilot's workload and greater flexibility of control laws will be possible. It is vital for the integrity of such systems that they be hardened against electromagnetic interference (particularly where powerful radar may be in use) and against nuclear electromagnetic pulse effects.

A "breadboard" version of the system, operating on a bench engine, has already given excellent results. Torque management, reversionary control, and operation of the self-monitoring system have been demonstrated. A prototype engine system is due to be bench tested in mid-1981. Testing in a tethered aircraft should take place at the end of the same year.

5.3 Life and Methods Programme. The life and methods programme at Rolls-Royce Ltd has been partly supported by MOD(PE) and the UK Department of Industry (DOI). It is aimed at improving the methods of design, stressing, and establishing safe lives of critical engine parts, whilst reducing the need for the expensive cyclic testing used at present on components of new engines. The programme covers gas turbine engines in general, and not just helicopter engines. The aims of the programme come under three headings:

a. Design of discs for improved low cycle fatigue life. Objectives include:

(i) refinement of finite element analysis methods, including the effect of creep and plasticity

(ii) fracture mechanics and crack propagation analysis to support "on-condition lifing" (sometimes referred to as "retirement for cause")

(iii) improved disc temperature estimation, and evaluation of disc material properties throughout their temperature ranges.

b. Turbine blade stressing. Objectives on turbine blade design include:

(i) methods of three dimensional temperature estimation combined with aerodynamic and cooling performance, which will be related to blade lifing and stressing programmes,

(ii) blade materials evaluation,

(iii) obtaining a measure of failure criteria from analysis of Service flight plans.

c. Engine structural dynamics including analysis of bladed disc models for vibration analysis of fan and compressor stages.

5.4 Cost Effective New Technology. The cost effective new technology (CENT) programme which has also been supported by MOD(PE) and DOI is being carried out by Rolls-Royce. The programme is aimed at improved design and the use of new manufacturing techniques in order to reduce cost, weight and fuel consumption. Many of the individual items in the programme are intended to fulfil more than one of these aims. Important items in reducing cost and weight include work on cast rotating parts such as discs and centrifugal impellers, improved gearing, and cast diffusers. Weight reduction items include improved oil cooling, gas bearings, gearing in composite materials, and work on ceramic components. SFC reduction is being sought through work on high temperature components and gas bearings.

6. ESTABLISHING TRADE-OFFS

6.1 Performance Trade-Offs. A helicopter designer is mainly interested in four performance criteria from an engine:

a. Low cruise (part-load) SFC

b. Maximum available power

c. Good handling

d. Low installation weight

A major problem with gas turbines has always been that of achieving a low cruise SFC, whilst at the same time having a satisfactory Contingency power level. This is illustrated in Figure 3. Figure 4 shows operating times at each power level for a typical helicopter. This information was obtained using an Engine Usage and Life Monitoring System (EUMS Mk 2), an MOD supported programme. It may be clearly seen that helicopter engines spend most of their operating time in two distinct and separate power bands: cruise and take-off/landing; the Contingency, or One Engine Inoperative (OEI), power levels are rarely used. This being so, improvements in cruise SFC could be obtained either if the power available from a smaller, lighter engine could be increased even for a short period of time (to level C on Fig 3), or if the take-off power required from the engine could be reduced (to level A), still leaving power in hand for OEI operation. The former approach, because it uses a smaller engine for a given power level than would be permitted using conventional rating structures, would require an Emergency power above the current Maximum Contingency level (see para 6.2). For a given twin engined helicopter a reduction in cruise SFC of 15 μg/J (.1 lb/SHPhr) over a 3 hour mission at an average of 350 kW (500 SHP) per engine is the equivalent of a reduction in weight of 130 kg (300 lb) in engine installation weight. Such a reduction might be achieved through the use of a flexible rating system. The second approach would require the use of multiple engines.

Figure 5 shows typical power levels required for 1, 2 and 3 engined versions of a helicopter. In the single engine case if engine power is lost the helicopter will have to auto-rotate and land. In the 2 and 3 engine cases it is expected that safe flight can continue on the remaining engines. To achieve the power range required for the helicopter, a two engine solution requires a much greater power range from each engine than a single engine solution. More importantly, it can be seen that a three engine helicopter requires a considerably smaller power range than does a two engine helicopter.

There are of course other aspects of 3 engine installations to be considered, such as flight safety, the intake design of a middle engine, and the engine/transmission configuration.

6.2 Power Versus Life. In very general terms engine life consumption can be measured in 3 ways. These are stress cycles, time/temperature history, and hours. The first is a measure of low cycle fatigue damage, the second is a measure of such damage as creep and thermal fatigue, and the third is a measure of "wear-out" damage. High temperature material limitations and design practice have been such that turbine blade failures have been more predominant than low cycle fatigue (LCF) disc failures, whilst wear-out failure modes are independent of these. This has meant that power versus life (in terms of time/temperature damage) has become a classic gas turbine trade-off.

The power levels that can be used depend on the nature of the Type Approval or qualification tests which the engine must undergo. The endurance tests called for by the major airworthiness

requirements are severe in terms of creep, to ensure durability and integrity. Supplementary tests are required to validate components subject to low cycle fatigue. The engine usage spectrum on qualification tests is quite different from that in operational service, in order to ensure that the engine has a severe test in a short time.

There is also a wide variation of creep and fatigue usage from engine to engine in an operator's fleet. The authorized engine or module life, currently declared in hours, is set so as to ensure that the most severely used engine does not suffer premature failure. It follows therefore that the majority of engines do not achieve their potential lives. Considerable progress has been made in recent years towards "on-condition" maintenance, but this concept is at present difficult to employ on parts subject to creep and more so on parts subject to LCF. The use of on-board recorders to monitor life in terms of creep and fatigue on individual engines would enable them to achieve a much higher proportion of their potential lives, and so save on engine support costs. Additionally, continuous monitoring of engine usage would permit a much more flexible use of higher engine ratings, as described in para 3.5, (Refs 6, 7, 8) and the use of Emergency power levels higher than current Maximum Contingency levels. Such Emergency power levels would be acceptable, provided that the integrity of the engine had been demonstrated during Type Approval testing, and the life consumption during their use was recorded. The Engine Usage and Life Monitoring System, currently being developed under MOD(PE) contract, provides a means of achieving these aims.

MOD(PE) are currently carrying out a review of their engine Type Approval requirements. The advantages to be gained from new philosophies, and changes necessary in the testing procedures are being considered (Ref 6,8). Whilst it is difficult at the time of writing to forecast the outcome of the review of our requirements, a number of factors can be mentioned.

In order to improve the simulation of the operational low cycle and thermal fatigue usage it has become the practice to carry out "Accelerated Mission Tests" (AMT) or "Sortie Pattern Tests" on development engines. These tests are aimed at reproducing as accurately as possible the damaging parts of the engine's operational usage. Although these tests are not currently a mandatory part of UK Type Test requirements, they have been extensively used. The possible role of the AMT in the qualification process is now being considered.

Emergency power levels are likely to differ, both in power and time of application, not only from engine to engine but in different applications of the same engine. The approval of Emergency power is therefore likely to require a flexible approach by the approving authority, rather than a single fixed test schedule, but nevertheless should be an integral part of the engine endurance testing.

The operational life of an engine is very much dependent on the environment in which it operates. The environmental tests which an engine and its accessories must undergo are therefore also being reconsidered.

6.3 Engine Power Off-Takes and Accessories. Engine auxiliary power off-takes and engine mounted accessories provide a simple reliable installation. These advantages have been eagerly seized by aircraft manufacturers. It may be, however, that changes in current design practice will occur.

With a free turbine engine it can be shown that for a given fuel flow every 1 kW taken from the gas generator may reduce the power turbine output by about 2-3 kW at the cruise condition. In practice this is seen as an increase in fuel flow. It would be preferable to drive as many engine accessories as possible from either the power turbine shaft or the helicopter gearbox. Accessories that could be considered in this light are engine oil coolers, generators and hydraulic pumps for the rotorhead which together typically consume 10-40 kW.

Ease of starting has important repercussions on the operational capability of a helicopter. As an example, both the type of starting system and its installed weight are directly related to the starting assist/resist characteristic of the main engine. An examination of engine specifications reveals that self-sustaining speeds and starting torques have gradually risen so that some modern engines require starting systems several times more powerful than those of similar size engines of 10 to 15 years ago. Even if cost analysis shows that this is an acceptable consequence of the trend towards decreasing fuel consumption in modern engines, the point may have been reached where unconventional starting systems may be worth considering. These would need to have a much higher power to weight ratio than current systems. Possibilities worth investigating, if the logistics aspects are acceptable, are hot gas starters or even two stroke piston engines.

In addition to the approaches mentioned above, the general use of standardized higher speed or direct drives would also lower transmission weight for both accessories and airframe. This approach has been examined periodically in the past, but present day requirements may motivate more determined development programmes and lead to a change in design practice for engine-mounted accessories.

7. TOWARDS A RANGE OF ENGINES

In this paper it has been shown that military helicopters, and hence their engines, fall into two general categories, long endurance naval types and short endurance battlefield and support types. The principal characteristics of each type depend on the relative importance, already outlined in Table 1, of factors such as specific fuel consumption, ability to function in the operational environment, and weight. Given a choice of several engines, all of which meet the technical requirements, the

procuring agency would ideally select the option with minimum life cycle costs although other aspects such as availability and industrial considerations would also influence the decision.

As discussed in para 3.4, minimum life cycle cost for the land-based short endurance helicopter might be achieved with an engine of low first cost, although with a higher SFC. However, for an engine to be used in a long-endurance naval helicopter, or in a combined fleet, the more sophisticated engine with lower SFC may be operationally necessary despite higher procurement cost.

It is in the interests of manufacturer and military procurement agency alike to minimise unit cost by spreading first costs over a long production run. In practical terms, this means developing an engine which will be competitive in as many markets as possible, both military and civil, even if the initial development is paid for with military funds. To attract commercial customers trade-offs may need to be modified to suit their needs. To increase sales volume further, several applications of the core engine should be possible, eg turboshaft, turboprop, turbofan, jet and industrial. Only after all potential markets have been assessed, should an engine manufacturer commit himself to full development of a new engine.

Military helicopters may broadly be categorized as light, medium/heavy, very large and anti-tank. With the exception of anti-tank and very large helicopters, each category defines a basic size of helicopter, which depends on the number of crew and passengers it has to carry. In the increasingly complex military environment, it is likely that the weight of equipment to be carried will increase, causing a tendency for the weights of most categories of helicopter to increase. Although power to weight ratios for naval helicopters are unlikely to change, some increase in installed power may be necessary for land-based types to take account of intake and exhaust protection. For anti-tank helicopters, power to weight ratios are likely to increase more markedly because of the need for agility. On the assumption that higher Contingency powers will be possible it is likely that engines with Take Off powers only slightly higher than those now in service will be adequate for most categories of helicopter. In the very large category, engines of powers in excess of 3000kW (4000 SHP) may be required. Due to the relatively small numbers of such helicopters their needs should be considered in conjunction with those of turbo-prop aircraft, if economic engine programmes are desired.

Whenever an engine is selected, it is important that an allowance is made for an initial margin of power to cope with the usual upward drift of weight during helicopter development, and for weight growth in service. The engine should also have stretch potential sufficient for later variants of the basic helicopter. This must be borne in mind at the conceptual stages of any new engine design.

8. IN CONCLUSION

The two most important goals for future military helicopters are likely to be improved environmental resistance and reduced cost of ownership. Research and technology programmes over the next ten years will continue to be directed towards these ends.

Although still important, less emphasis than hitherto may be given to improving performance. Indeed, such improvements in helicopter performance as might be gained from the engine and airframe may be partly offset by the measures taken to achieve other goals.

The introduction of engine condition monitoring systems should make an impact in the areas of engine rating philosophy, which may allow smaller, lighter engines to be used; and should reduce support costs through improved "on condition" maintenance techniques.

Reappraisal of accessory mounting practice may lead to small but useful improvements in efficiency.

Engine qualification requirements need to be continuously reviewed, both to ensure that new engines will be operated satisfactorily in a changing military environment, and to take advantage of revised rating structures made possible by improvements in engine condition monitoring.

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TABLE 1

REQUIREMENT	ENVIRONMENT					
	Battlefield Army/Air Force			Over Water Navy/Air Force		
	Priority			Priority		
	1	2	3	1	2	3
Resistance to the Battlefield Environment: FOD, IR, Noise, Battle Damage Survivability	√					√
Resistance to Icing and Salt Corrosion			√		√	
Operating Range and Endurance			√	√		
Agility and Engine Handling	√				√	
Maintainability		√			√	
Life Cycle Cost		√			√	
Flexible Rating Structure			√			√
Low Weight		√			√	
Multi-Fuel Capability			√			√
Improved Starting			√			√

Priority 1 Requirements essential to the performance of the intended mission of the helicopter.

2 Important requirements, but less crucial than those in category 1.

3 Desirable features, if they can be achieved without prejudice to items of priority 1 and 2.

FIG 1 VARIATION OF SPECIFIC POWER AND SPECIFIC FUEL CONSUMPTION WITH CYCLE PRESSURE RATIO AND TEMPERATURE

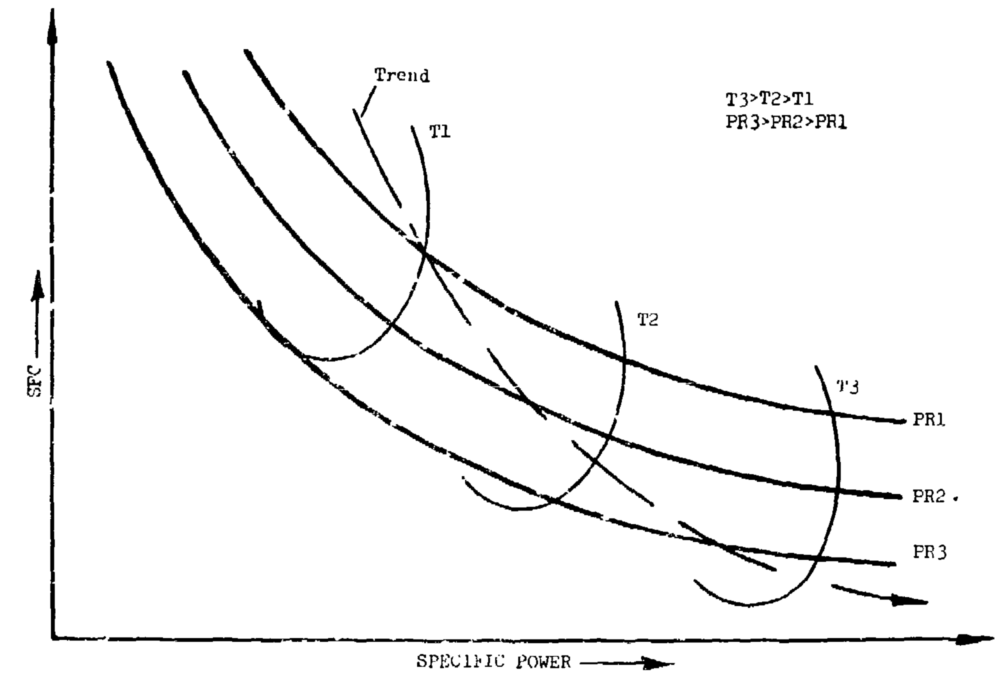

FIG 2 DISTILLATION CURVES FOR TYPICAL FUELS

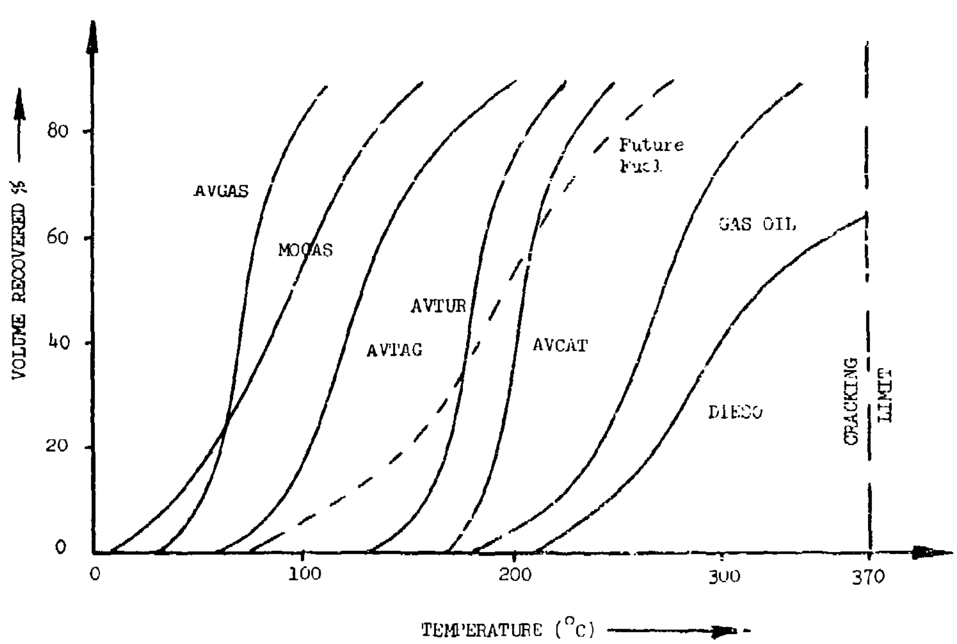

FIG 3 ENGINE SIZE EFFECTS

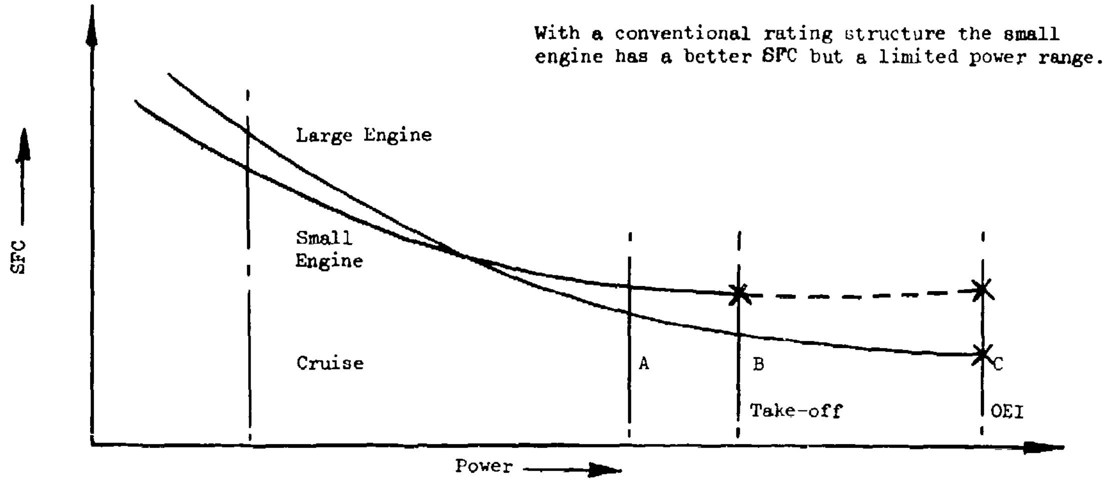

FIG 4 TYPICAL TIME/TORQUE ANALYSIS

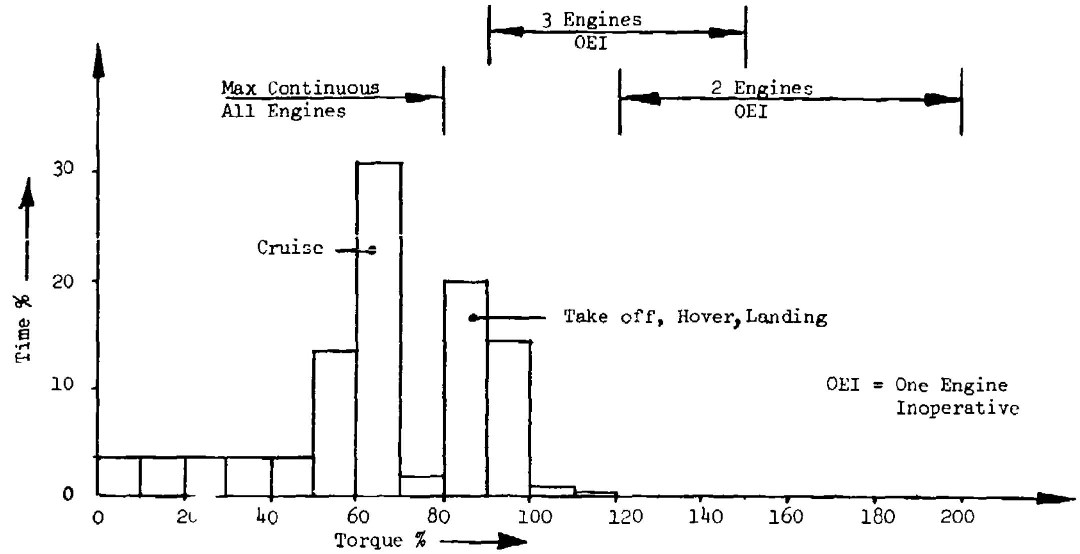

FIG 5 TYPICAL POWER REQUIREMENTS FOR MULTI-ENGINED HELICOPTERS

HELICOPTER POWER	% ENGINE POWERS			
RATING/POWER LEVEL	%	1 ENGINE	2 ENGINES	3 ENGINES
Cruise	60	60	60	60
Max Continuous	80	80	80	80
Take-Off	100	100	100	100
Inter Contingency	60	-	120	90
Max Contingency	75	-	150	112.5
Emergency	100	-	200	150

DISCUSSION

C.Albrecht, US

Could you elaborate as to how you might exploit the potential life of life-limiting parts?

Author's Reply

There are two possible approaches: first, to reduce the life lost during overhaul when components are removed because they have less residual life (in hours) than the customer is willing to accept; second, to extend the life of parts that have not been subjected to severe conditions. Both approaches require the use of an engine usage and condition monitoring system (EUMS).

K.Rosen, US

Did you imply in your discussion that engine cost is not an important parameter?

Author's Reply

Prime consideration, in our view, is that of total helicopter costs, and not just engine costs.

MECHANICAL ADVANCES IN THE DESIGN OF SMALL TURBOSHAFT ENGINES

by

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ABSTRACT

Mechanical components have a significant influence on the efficiency of a small gas turbine engine. Some of the performance losses associated with the design of the power transmission and internal air system are defined and discussed. Improvements in engine efficiency must be considered in conjunction with cost, reliability and size or weight. Many of the problems considered are applicable to gas turbines in general but become acute in small engines due to the adverse effects of scale on many component design parameters. To meet the increasing demand for more efficient powerplants the mechanical research engineer must improve the analysis of mechanical component behaviour to produce optimised engine designs.

INTRODUCTION

As a general rule the inefficiency of an engine due to mechanical considerations does not scale with engine size. Thus, for a small engine, the losses related to the mechanical systems will be a much higher proportion of the output power than is the case with its larger relatives (Figure 1). The problem is compounded by the requirement to produce shaft power rather than jet thrust thereby necessitating a low speed output transmission system. The mechanical control of the internal air system also becomes more difficult as size decreases because features such as the clearances of seals and blade tips do not scale directly.

The current emphasis on overall engine efficiency (1) has meant that the mechanical aspects of small engine performance have come under careful scrutiny. However, the overall requirement of cost, reliability, mechanical efficiency and compact design are highly interrelated and often incompatible.

The purpose of this paper is to review the progress in the mechanical systems of small engines and to assess the current state of the technology. The discussion is limited to the power transmission and internal air systems. These two between them limit the overall efficiency of the engine by incurring direct mechanical losses from the rotating components and cycle losses due to bleeds and leaks from the main engine gas stream.

POWER TRANSMISSION SYSTEMS

In the case of a gas turbine engine the term 'transmission system' encompasses all the components that support rotating shafts or transmit power. In a multi-shaft engine such as the R-R Gem (Figure 2) a large proportion of the engine volume is concerned with the transmission and its associated components such as the oil system.

Bearings

Over the past 20 years the main improvement in the life and load capacity of rolling bearings has been due to material changes; firstly the move from carbon chrome to tool steel and secondly the introduction of vacuum melted materials rather than air melted. Bearing lives for a given specific load have improved by an order of magnitude. However, there now seems to be little scope for further rapid changes and future improvements will be achieved by subtle refinements to design, materials and lubrication. The internal geometry in ball bearings is also approaching its limit. Very close conformities are required to withstand the high applied and centrifugal loadings yet at the very high speeds proposed for new engines the resultant large contact ellipses will give rise to considerable heat generation (to the detriment of the oil films). The requirements of high load capacity and good mechanical efficiency are incompatible and the added complication of high speed is pushing conventional technology to its limit (2).

In the case of roller bearings the difficulties stem from a combination of high speed and low loads. Under such conditions the high viscous drag on the cage and low frictional force between the rollers and driving race will result in skidding (3). Skid itself is not necessarily destructive; it is the combination of dynamic loads and thin oil films which, in the presence of slip, allows asperity contact and surface damage. Conventional methods of skid control involve some degree of preloading across

the bearing. In small engines this is unacceptable due to the high stiffness of the tracks. Research into the cause and nature of slip in high speed roller bearings (4,5 and 6) has shown that it is possible to control slip by a revised design of the bearing, lubrication system and housing. Both slip and heat generation are functions of the drag acting on the cage and rollers. Thus, any techniques that can be used to reduce it such as low oil flows and small rollers, will not only reduce slip but will also improve the mechanical efficiency. Unfortunately, such techniques require the bearing to be externally cooled and the overall system will become more complex.

In addition to internal heat generation, small engine bearing chambers receive a large amount of external heat input due to their proximity to the gas stream. Oil is supplied to a bearing chamber to achieve two major tasks - lubrication and cooling. Since a large proportion of the heat to be removed from the bearing chamber originates from outside the chamber it can be argued that only a small proportion of the cooling oil need enter the chamber and most of the external cooling can be achieved by means of an oil jacket, separately fed and scavenged. (Fig.3) Such a system would result in reduced churning losses and less chance of oil leakage since the total volume of oil is now much smaller compared to the volume of the chamber.

At the same time, attention should be given to reducing the radiation, conduction and convection heat transfer paths between the gas stream and bearing chamber.

In future engines it may be possible to replace the high speed, lightly loaded turbine roller bearings by self acting gas bearings. Such a system can run at elevated temperature with no oil flow and with very little power loss giving a simple and elegant design (figure 4). However, the load capacity of such bearings is low compared with the rolling bearings that they replace and they cannot yet replace highly loaded thrust bearings.

Gears

As is the case with bearings, the greatest improvement in gears has been the materials. If gears are to be run successfully at very high speeds then they must approach geometric perfection as it is often the internally generated dynamic loads rather than the direct applied load that is the cause of failures. Figure 5 illustrates the lubrication regimes for a pair of gears. At the elevated speeds encountered in small gas turbines boundary lubrication is not usually a problem. The gears will be designed to run under full lubrication (zone II) but additional loads generated by errors in tooth form or position can easily cause the operating point to shift to zone III, resulting in failure. Final breakdown may be in the form of scuffing (a rupture of the oil film,) surface pitting (rolling contact fatigue) or fatigue at the roots of the teeth. All of these can be controlled by careful manufacture of the tooth flanks, roots and pitch.

The introduction of conformal gears in some low speed aerospace applications has given an increase in load capacity due to the stronger tooth form but the higher sliding velocities and consequent difficulty in maintaining the oil film has prevented their use at higher speeds.

Any attempts to properly optimise the lubrication system reveals an incompatibility in lubricant requirements for bearings and gears. Clearly the entire transmission system must share one common lubricant but the ideal lubricant for gears has a high viscosity to maximise the oil film and a low traction coefficent to minimise the frictional losses whereas bearings require a low viscosity to reduce cage drag and a high traction coefficient to drive the cage with as little slip as possible.

Analytical Techniques

Since the introduction of small turbo-shaft engines there has been a tendency to neglect the analytical techniques as the transmission systems have become increasingly sophisticated. The result is that conventional prediction techniques are not capable of dealing with the subtleties of the new, high speed transmissions.

During the early 1960's reliable solutions to the elasto-hydrodynamic problem became available. This lead to improved estimates of bearing and gear life. Gear analyses are normally based on traditional techniques and individual experience, a method that generally works well. The analysis of high speed bearings, is, however, less reliable, and the techniques are constantly being developed. Many of the existing methods (7,8) involve simplifications that are no longer acceptable. A complete analysis of a bearing system is a complex undertaking (9,10) and the art is to simplify these techniques while retaining the necessary variables. An example of this is slip in ball and roller bearings where relatively simple techniques (2,11) have produced reliable results. It has become increasingly clear that the thermodynamics of the bearing chamber must be considered if a proper analysis of slip is to be achieved (figure 6). The introduction of oil jackets and other cooling techniques further complicate the thermodynamic model.

Internal Air Systems

The internal air system comprises a number of mechanical devices which control the airflows within the engines to govern the sealing of hot gases and bearing chambers, the temperature of 'hot' components and the thrust load acting on the mainshaft ball bearings. It is essential to perform each of these important functions with the minimum expenditure of air since any air bled from the compressor/turbine cycle will have a significant affect on engine performance.

Turbine Blade and Nozzle Vane Cooling

As TET's have been increased, new high temperature materials and complex cooling systems have been developed to achieve satisfactory component stresses and life. This has resulted in problems of mechanical design, for the components themselves and the air delivery systems where the requirement for high pressure air feeds to turbine blades and nozzle vanes exacerbates any leakage problems due to small engine seal clearance limitations. As the amount of air needed for cooling is dependent upon its delivery temperature the heat transferred to the cooling air feed should be kept to a minimum. The use of nozzles to preswirl the air in the blade cooling feed system produces a temperature reduction when transferring air from a static to rotating environment.

Since mainstream air bleeds are a debit to the engine performance all cooling flows should be minimised. This is especially true for nozzle end wall cooling flows due to the influence of film cooling air boundary layers on secondary losses in low aspect ratio nozzle vanes.

The physical size of small engine nozzle vanes and turbine blades creates difficulties in manufacturing the complex cooling passages needed for high effectiveness cooling especially in the leading and trailing edge regions.

It is therefore important to consider carefully any performance losses associated with the introduction of component cooling along with the temperature limitations of new materials when deciding upon the optimum TET for a new or uprated engine.

Disc Sealing and Cooling

In early gas turbines little emphasis was placed on preventing hot gases from entering the core of the engine. However, as speeds and temperatures have increased it has become necessary to prevent hot gas ingestion at the disc rim and avoid over-heating the highly stressed turbine discs. The technique that has been developed is to bleed relatively cool compressor air radially outwards across the face of the disc. Research into the sealing requirements of a rotor/stator cavity has shown that the minimum airflow needed is a function of rotational Reynolds No., disc radius and rim gap. (12).

The quantity of air used for turbine rim sealing should be the minimum possible since it has a double effect on engine performance. Firstly, air bled from the main gas stream does no useful turbine work and secondly, this air can cause severe disturbance to the turbine blade flows when injected into the gas stream at the disc rim. Improving the geometric design of the rim seal allows satisfactory hot gas sealing with a smaller airflow. For large gas turbine engines complex rim seals have become a standard feature but are not easily designed into small engines since it is difficult and expensive to manufacture reduced scale seals of this type to suitable tolerances.

Work is in progress to optimise the rim seal design using realistic small engine geometries from which a comparison can be made with the correlation for more idealised geometries (Figure 7).

Improved understanding of the complex flow mechanisms associated with rotating components has enabled the air system to be optimised for better control of component temperatures.

By varying component geometries and air bleeds it is possible to:-

a) Influence temperatures in turbine discs to alleviate thermal stress problems associated with adverse temperature gradients possible during rapid start/take-off operation. Figure 8 shows an example of an integral blade and disc where the disc rim responds very rapidly to changes in gas temperature.

b) Reduce the windage generated by the rotating components which, as well as being a direct mechanical engine power loss, is largely dissipated as heat. This latter effect can cause severe temperature problems in components such as centrifugal compressors operating at high rotational speeds unless the windage is controlled. (figure 9).

Turbine Tip Clearance Control

As faster hotter engines have evolved, blade stresses have forced the trend away from shrouded turbine blades (figure 10(b)) to unshrouded blades (figure 10(a)). With separate shrouds it is essential to keep the turbine tip to shroud gap as small as possible at cruise and take-off conditions so as to minimise overtip leakage and thereby avoid the associated reductions in turbine efficiency and engine power output. Comparison of achievable tip clearance with blade height highlights the problem of small engine tip clearance control where sheer size causes difficulties in manufacturing and operating a mechanically active tip clearance control system in an environment of 1100-1200°C and 45000 rpm.

The main problem with metal shrouds at high turbine entry temperatures (TET) is distortion and as TET's are increased, metal shrouds require cooling systems of ever increasing complexity to maintain satisfactory metal temperatures and control growths to give acceptable tip clearance. With such arrangements there must always be a compromise between minimum tip clearance at high operating powers and possible transient pinch points during acceleration or shutdown. These shroud cooling systems either use air which is ejected directly into the gas stream or air which is subsequently used to cool the nozzle guide vanes. The former system incurs a performance penalty and the latter system degrades the quality of the nozzle cooling air.

An insulating layer can be used on these shrouds to reduce the gas stream heat transfer and should ideally possess abradable properties to prevent blade damage during a transient rub. It is however difficult to develop a satisfactory abradable material which can withstand gas erosion at elevated TET's,but recent research has produced new more suitable materials.

An alternative to the complexity of cooled and insulated shrouds is to use a high temperature ceramic material such as silicon carbide (figure 11). Certain ceramic materials have low expansion properties and excellent high temperature capabilities enabling a ceramic shroud to operate uncooled at high TET's with low distortion, small clearances at high powers and without transient blade rubs.

The disadvantage of ceramic shrouds is that ceramics are relatively brittle compared to current high temperature metal alloys and therefore any designs should ensure that ceramic components are not subjected to severe tensile loads, either self generated thermal shock or applied physical load. Care should be taken to mechanically insulate any ceramic component from adjacent metal structure by using a compliant mounting system to reduce any thermally induced stress concentrations.

Turbine shrouds are an ideal application for ceramics since they are not highly stressed, yet the potential gain in turbine efficiency is substantial.

Bearing Chambers and Loading

The combination of turbine reaction loads and air pressure loads on disc faces can result in net aerodynamic loads in excess of 5000N, acting axially on an engine shaft. To achieve adequate bearing life these loads are generally reduced by incorporating balance piston discs with air pressure differentials between their faces separated by a labyrinth seal. The characteristics of a labyrinth seal are such that the leakage flow is a function of the pressure difference and the operating clearance. Seal clearances are governed by tolerance build up, thermal and centrifugal growths and radial clearances in the bearing and squeeze film. These factors are similar for small and large engines which often create small engine problems where seal airflows are in excess of those required for rim sealing bearing chamber sealing or cooling purposes.

It may be necessary to incorporate bearing chamber vents to maintain satisfactory seal pressure drops and prevent oil leakage. As well as wasting expensive compressor air, the presence of large quantities of relatively hot air (compared to oil temperature) in the bearing chamber produces increased bearing temperatures and places greater demands on the oil cooling system. To improve sealing efficiency new types of seal are being considered such as brush seals; hydraulic seals which can theoretically withstand 150 psi at small engine sizes and speeds, and self acting lift off seals which operate on the same principle as gas bearings. Positive bearing chamber seals would reduce air wastage and heat rejection to the oil and would allow suction scavenge to remove the oil as opposed to pressurised bearing chambers.

As well as reducing the applied bearing thrust loads progress has been made in increasing the load capacity of the bearings. However, higher load capacity results in larger bearings with increased power losses, especially for multi-row bearing arrangements. A design compromise must therefore be reached between mechanical loss and air system bleed performance penalties.

CONCLUSIONS

This paper has discussed some of the difficulties encountered in providing satisfactory mechanical systems for modern small, high speed, high temperature, turboshaft engines. The increasing demand for more efficient engines requires the mechanical components to be optimised to a much higher degree than has previously been necessary. New analytical methods are needed if that optimisation is to be achieved as the engine is designed.

Since the introduction of small gas turbines considerable improvements have been made in the power transmission and internal air systems in respect to both component efficiencies and lives. However, as development continues the law of diminishing returns inevitably applies thus further improvements present a challenge to the research engineer and require courage on behalf of the designer, who must include ideas that are, in some cases, totally new.

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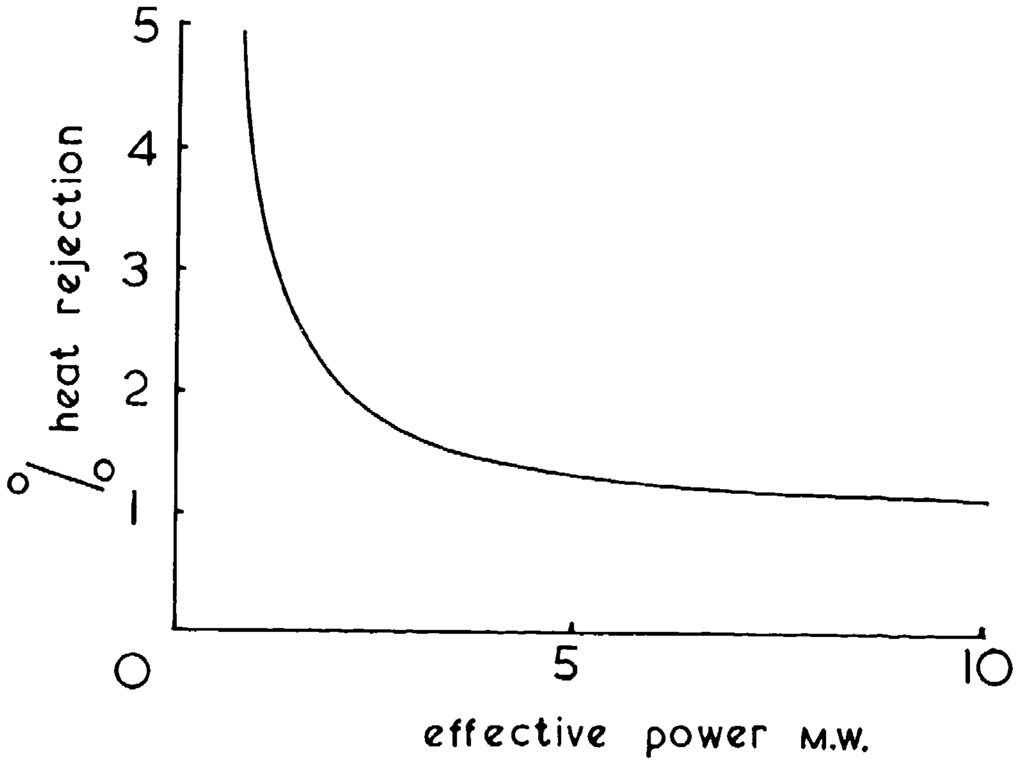


Fig.1 Relationship between effective engine power and heat rejection to the oil system in aircraft gas turbine engines

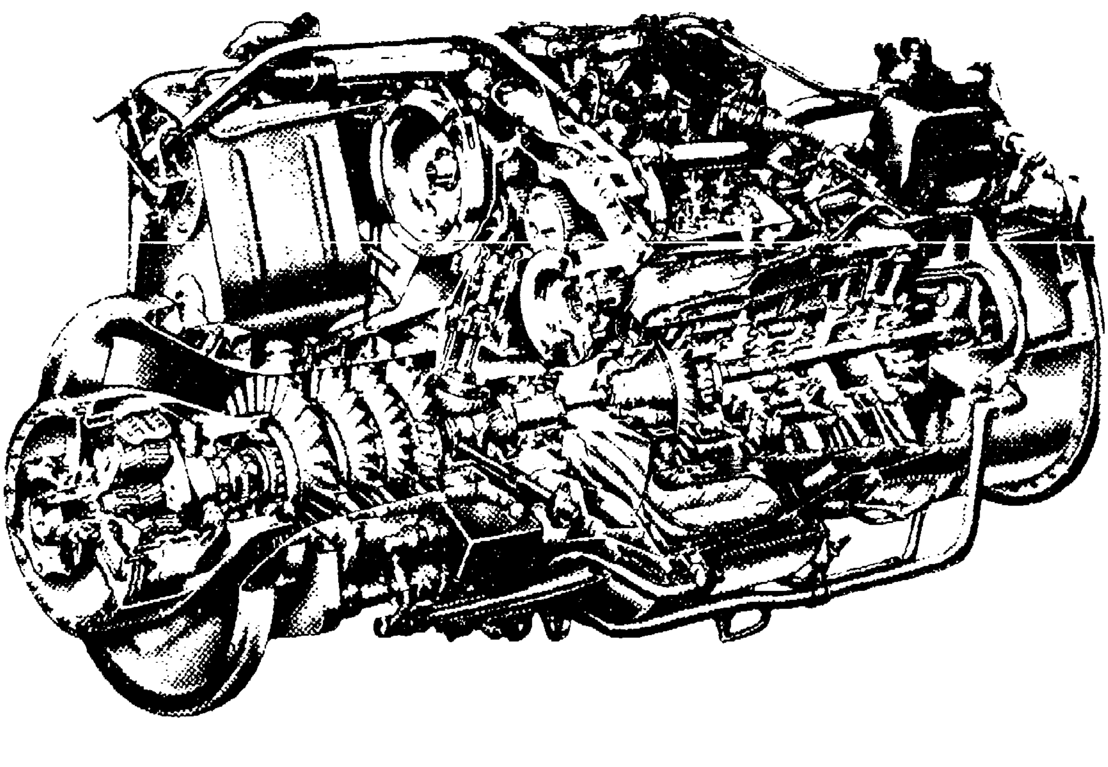


Fig.2 The Rolls-Royce Gem 2 three shaft (700 KW) gas turbine engine

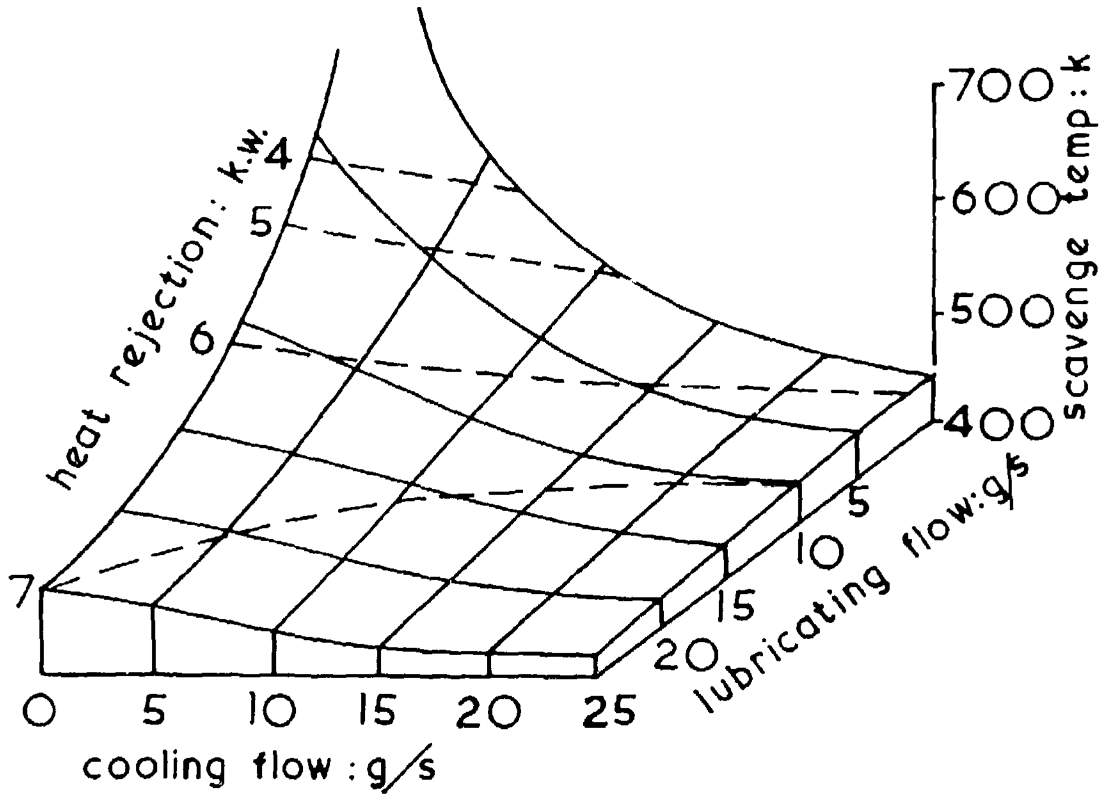


Fig.3 Heat rejection to the oil system as a function of cooling and lubricating flows for a typical externally cooled high speed turbine roller bearing chamber

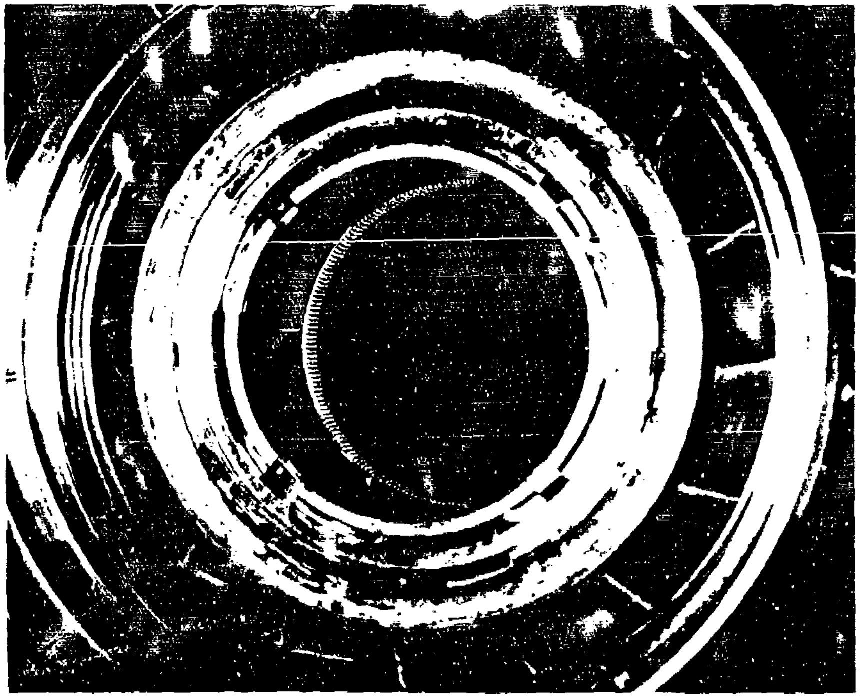


Fig.4 Experimental gas bearing mounted in the Gem low pressure turbine nozzle

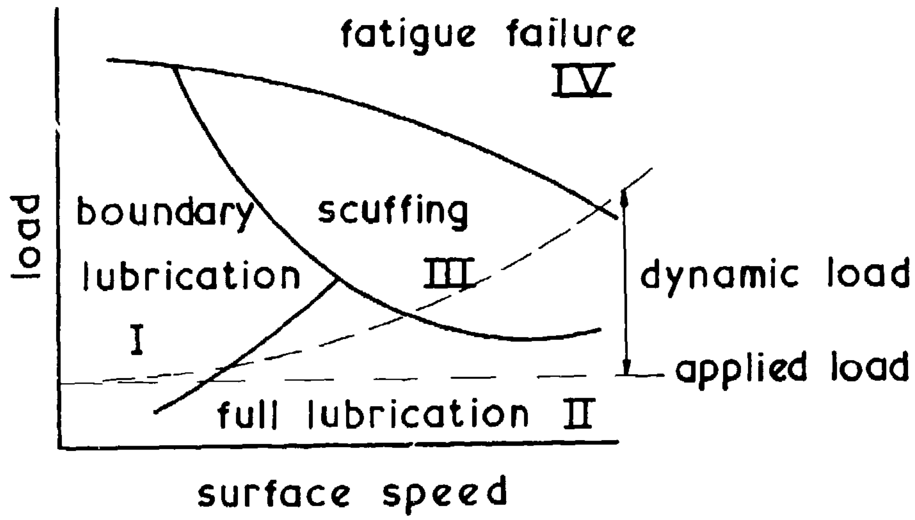


Fig.5 Lubrication regimes for a pair of gears. As the speed rises so do the dynamic loads and the operating point may move to a failure mode

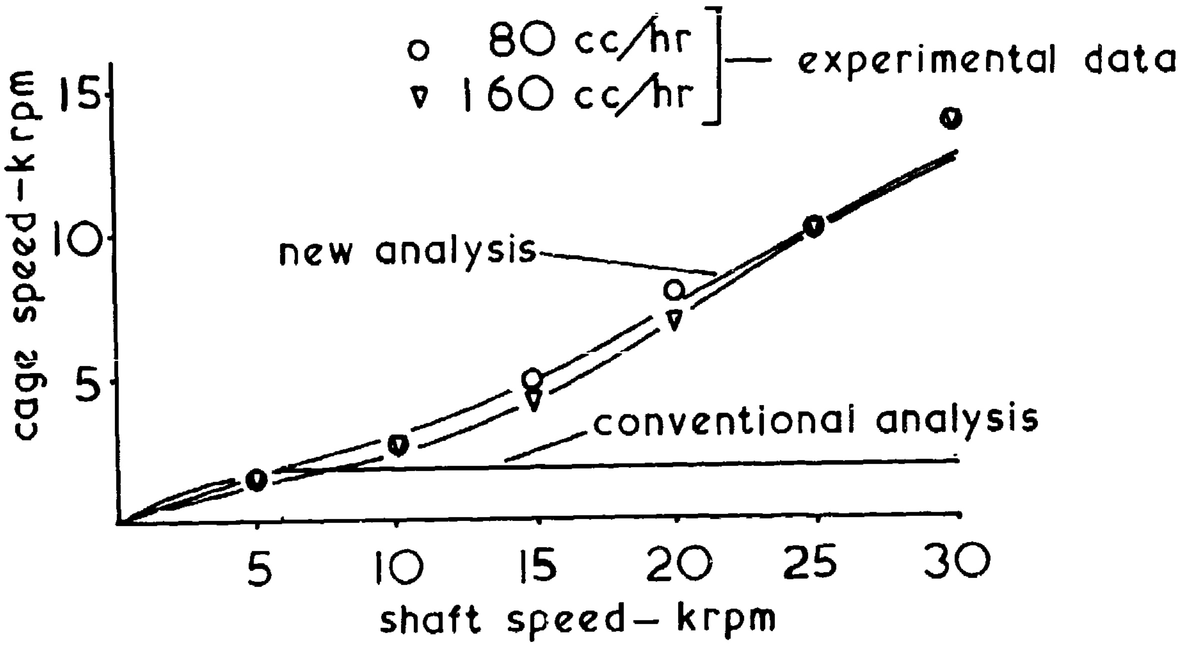
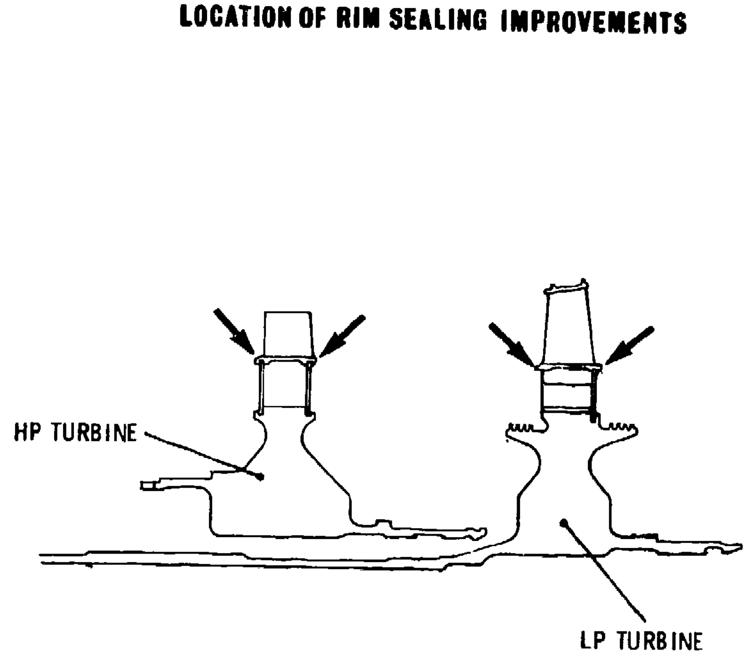


Fig.6 Comparison of new and conventional cage slip analyses for a typical small gas turbine roller bearing



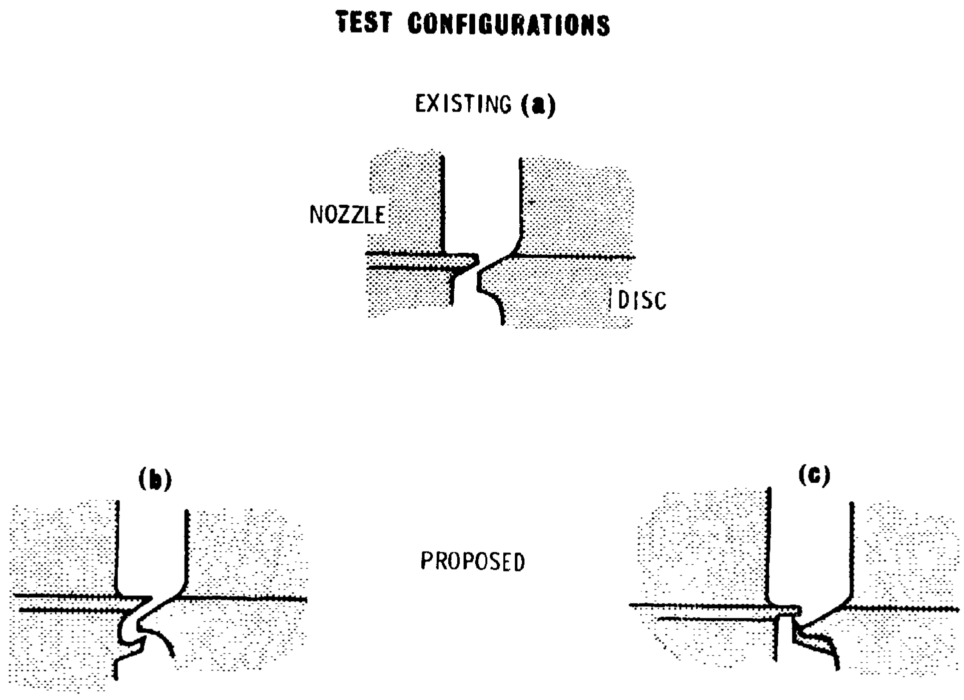


Fig.7 Experimental turbine disc rim seals

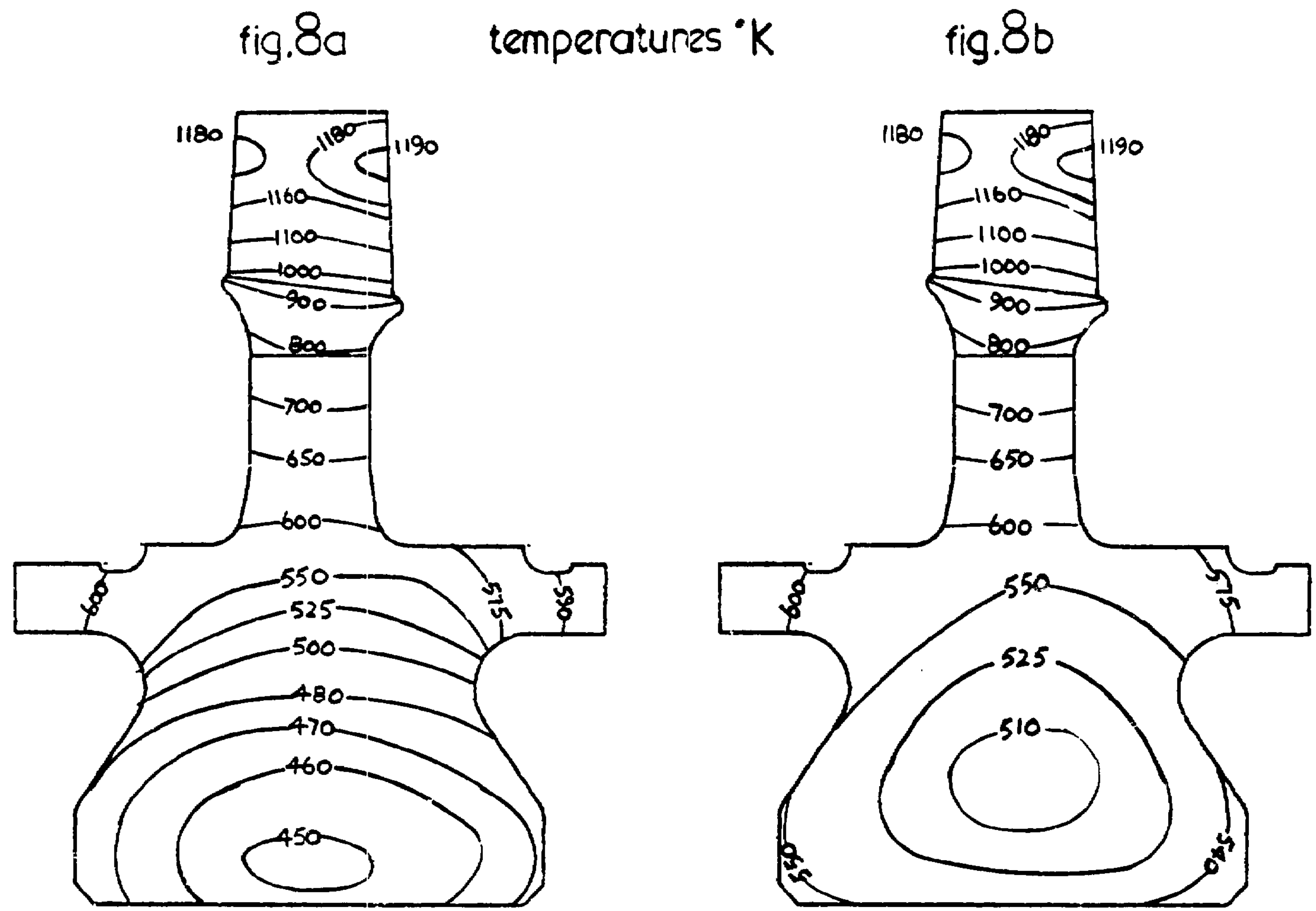


Fig.8 Typical take-off transient disc temperatures for the RB318 turbine (a) with reduced bore heat transfer and (b) with increased bore heat transfer

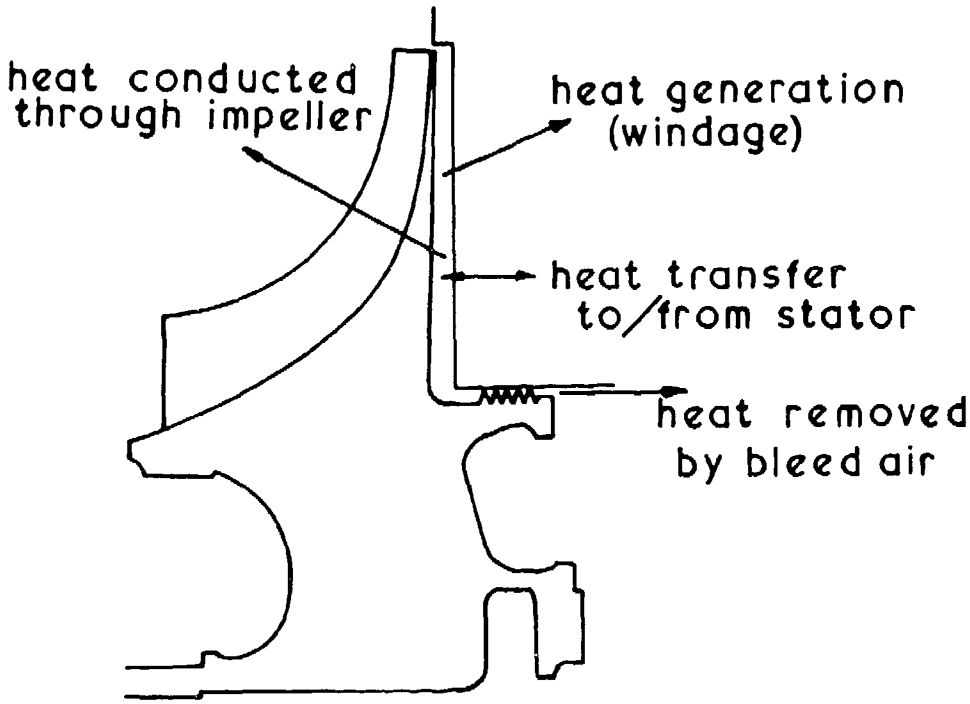
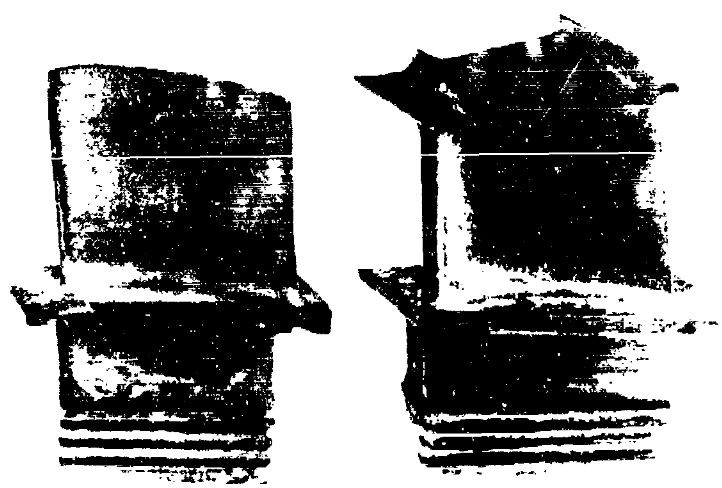


Fig.9 Heat transfer mechanisms in a centrifugal compressor



(a) (b)

Fig.10 High pressure turbine blades (a) without integral shrouds and (b) with integral shrouds

(a)

(b)

(c)

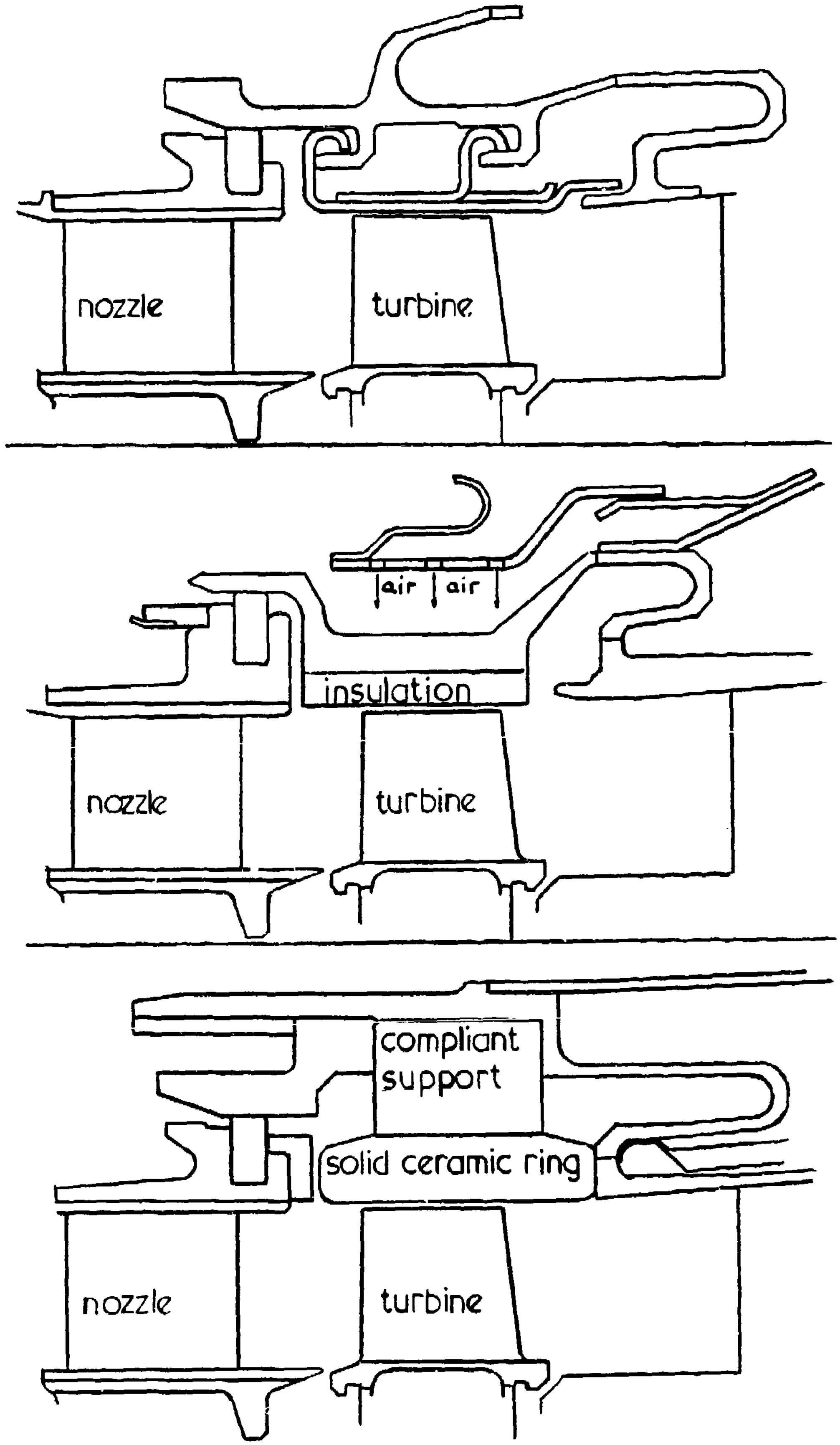


Fig.11 Examples of turbine blade shroud arrangements
(a) uncooled, (b) cooled and insulated and (c) ceramic

DISCUSSION

B.Shotter, UK

Could the authors comment further on their suggestion that conformal gearing is not appropriate for high pitch line speeds? My experience is contrary to this statement. These gears can operate up to about 100 m/s and perhaps even further. As far as has been investigated, the lubrication losses with conformal teeth have always been lower than with the equivalent involute teeth.

Author's Reply

We thank you for your comments and are interested in learning more about your recent developments in this field.

D.Hennecke, Ge

In Figure 11 you show turbine shroud cooling concepts. What insulation material do you use in design (b)?

Author's Reply

The insulation in design (b) is a nickel-aluminium filled honeycomb.

D.Hennecke, Ge

Have you already built a ceramic shroud as shown in design (c) and what is your experience?

Author's Reply

A ceramic turbine shroud will be tested in the Gem Advanced Mechanical Engineering Demonstrator (AMED) in September 1981.

D.Hennecke, Ge

How do you handle rub-in that is desired during running in of the engine?

Author's Reply

The rubbing characteristics of ceramic materials are being evaluated in a comprehensive research program to support the implementation of ceramic components into development engines.

W.Heilmann, Ge

Is not the problem of high rate of heat to oil related to your design (many bearings, seals, heat conduction through struts, etc.)?

Author's Reply

In a three-shaft engine such as the Rolls-Royce Gem one has to pay somewhat for the higher engine efficiency and response rate with an increase in mechanical losses due to the large number of bearings and seals required. To offset this mechanical loss penalty, we are incorporating the results of our research into the behavior of mechanical components leading to improved bearing chamber designs.

K.Rosen, US

Have you considered a customer bleed valving system that would allow air to be taken off the high compressor at low-speed operation and low compressor at high-speed operation?

Author's Reply

Provision of alternative sources for customer air bleed at different engine powers could produce small but significant gains in engine cruise sfc and would be relatively easy to incorporate in our two-shaft gas generator arrangement. To evaluate this sfc improvement in detail for particular applications and assess the cost effectiveness of such a system will require information on pressure, temperature and mass flow requirement for the cabin bleed at all operating conditions.

D.Griffiths, UK

You spoke about the benefits to be obtained in turbine efficiency by close control of tip clearance. Is active tip clearance control applicable to helicopter engines and have you considered such a system?

Author's Reply

Active tip clearance control could give improvements in turbine efficiency in helicopter engines, although it is more difficult to install such a system in a small engine, which has a reverse flow combustion chamber, due to space and temperature limitations.

ADVANCED COMPONENT DEVELOPMENT DESIGN BASIS FOR NEXT GENERATION MEDIUM POWER HELICOPTER ENGINES

by

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SUMMARY

A next generation medium power helicopter engine will have to show significant advances, when compared with engines representing todays high standard of technology. Towards this end, reduced fuel consumption will be a prime contributor, but new development can only be justified, if it results in improved overall engine economics as well as reliability.

The basis for success is the application of advanced but proven component technology. Design activities need to start well in advance of the actual engine program to define the environment for component development, with the target to have technology and design methodology available at the beginning of engine development.

Starting off with an optimized thermodynamic cycle, it will be discussed which components need intensive research and development to render a mature and reliable first production engine. Present date results, with regards aerodynamic and structural technology, are presented from both rig and demonstrator test programs.

SYMBOLS

M_C	cooling air mass flow
M_F	fuel flow
M_{RED}	reduced mass flow
M_1	compressor mass flow
N	rotor speed
P	power
T_{C1}	cooling air temperature
T_G	gas-temperature
T_{t3}	stator outlet temperature of the gasgenerator-turbine
T_W	wall temperature, local
η_C	compressor efficiency
η_{CE}	cooling effectiveness
η_{GGT}	gasgenerator-turbine efficiency
η_{IS}	isentropic efficiency
π	pressure ratio
π_C	compressor pressure ratio

1. TARGETS FOR DEVELOPMENT

Helicopters today are an established part of aviation. Towards this success the high standard of technology imbedded in present generation engines has paid its vital contribution. Against this background, and considering the ever increasing cost of development, two questions will be rightfully asked

- with current engines'performance being so far advanced, what further significant improvement is there to be realized, and
- what is the impetus, that has the industry working on component development for a next generation helicopter engine today.

Fig. 1 is an examplatory attempt to give part of this answer. It compares the levels of specific consumption over Take-Off-Power, that is the performance offered today with the targets for a next generation of engines. This improvement reflects more ambitious thermodynamic cycles matched to the aerodynamic and mechanical quality of advance technology components (Ref. 1, 2, 3). From that figure it also becomes apparent, that the range around 900 kW is not yet adequately represented.

However, reducing SFC cannot be the sole justification for new development. In spite of increasing fuel cost, it remains but one important aspect of engine economics, in which further reduced weight and volume, higher reliability together with better maintainability figure equally as high. It is this requirement for better system performance and overall economics, resulting in reduced operating cost, that the industry is reacting to, when proposing a next generation of engines.

The basis for success still remains the application of advanced components in terms of aerodynamic and mechanical design. Considering the tightening schedules available for engine development, component development needs to start well in advance. Testing in both rig and demonstrator are paramount, in order to have proven technology and design methodology ready at the beginning of the actual engine program.

The work discussed here is oriented at the medium power helicopter, specifically the 900 kW engine class. A major portion of it being MTU's contribution towards a program called "MTM-Technology-Demonstration". It is jointly pursued with TURBOMECA and in part supported by the competent authorities in France and Germany* (Ref. 4).

2. COMPONENT DEFINITION AND CYCLE SELECTION

Early design activity is called for to define the environment for component development. Towards that end a base line engine design, properly reflecting the afore mentioned requirements, has to be established. This will then put advance technology targets into proper perspective, and allow for integrated component activity.

Recognizing the small size of future power plants, particular attention has to be given to the selection of an optimum cycle (Fig. 2). Aside from the components'technology content, limitations have to be accepted, which result from turbine cooling and size effects (Fig. 2a). To better point out these aspects, turning the same correlation inside out, though less common, is more convenient (Fig. 2b). Plotted are turbine-stator outlet temperature versus pressure ratio, with specific fuel consumption and specific power used as parameters. This to demonstrate, that two important effects need be taken into account:

- high stator outlet temperatures tend to increase cooling air requirements, leading to deminishing returns in cycle efficiency
- coupled with such temperatures, there is an increase in specific power. The resulting reduction in turbomachinery size, augmented by higher pressure ratios, wants careful trading against the now gaining negative influence of secondary losses on component efficiency.

A design diagram (Fig. 3) reflects these inputs and indicates, that for a given engine size in terms of power, the range of design parameters to achieve optimum specific consumption is rather narrow. Extending the analysis to partload performance, as well as mission fuel consumtion arrives at the same results (Ref. 5).

The requirement for optimum partload performance naturally affects component design. Especially compressor and powerturbine have to render high efficiencies at low aerodynamic speeds.

On the way to component definition additional factors of importance have been considered

- high gasgenerator aerodynamic overspeed capability and temperatur margin to cover emergency power ratings
- short acceleration time, requirering a low inertia rotor, sufficient surgemargin for the compressor and a gasgenerator-turbine with high efficiency at reduced aerodynamic speeds.

Within that framework the base line engine has been defined (Fig. 4), incorporating the highest potential for lowest specific consumption and a life of 6000 mission hours. It consits of a variable geometry axial/radial compressor, a reverse flow combustor, a two stage gasgenerator-turbine and a two stage powerturbine with front drive. This sets the environment for component work

- basic development in the rig, and
- substantiation of the technology level achieved under actual engine conditions in the Gasgenerator Demonstrator (Fig. 5),

out of which but a few selected aspects will be presented.

3. GASGENERATOR-TURBINE

Targets for development are

- high performance with careful attention to minimizing secondary effects on turbine aerodynamics, such as tip clearance control and introduction of cooling as well as leakage air into the main gas stream (Ref. 6).

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* in Germany the Federal Ministery of Defense (BMVg)

- Structural design to satisfy life-, reliability- and cost requirements in an environment of elevated thermal- and mechanical loading.

Fundamental aerodynamic development is being carried out with the aid of a cold rig (Fig. 6), designed to permit separate variation of the individual cooling air flows, i.e. to first stage stator, first stage blade and liner, and second stage stator. Aside from verifying, that target efficiencies have been comfortably achieved, these tests have yielded very important answers, which will help to further improve the component (Fig. 7): Cooling airflow of the first stator effects capacity only, a factor when properly matching the gasgenerator. However, both first blade and second stator cooling air effect efficiency. Even though already moderate, this penalty can be further reduced by optimized cooling air reintroduction.

The amount of cooling air required has a detrimental influence on cycle efficiency, as has been indicated before. Selection of high thermal strength material for the airfoils, together with good cooling effectiveness becomes of prime importance for structural design. Testing in the demonstrator (Fig. 8), using thermal paints, has made evident that cooling airflow can be reduced. For example on the first stage vane leading edge, temperatures of the hottest airfoil were within target, the trailing edge was cooler than design. Similar results were obtained for the other blades and vanes.

The structural design of the first stage rotor is clear cut, separate cooled blades held in a powder metal disk; this technology by now being well established at MTU. The second rotor (Fig. 9) becomes more of a challenge, in terms of cost effectiveness. With no blade cooling required, an integrally cast wheel is feasible, but marginal with regards cyclic life. Here diffusion bonding presents an efficient technique to combine materials, casting for high thermal strength of blades and rim, powder metal for excellent low cycle fatigue life of the disk. As of today, structural rig tests have met all expectancies and this design is on the way to prove its potential in the demonstrator.

4. COMBUSTOR

The combustion chamber is a key component for a successful engine. In view of elevated turbine inlet temperatures, its own durability as well as its performance's influence on the life of downstream components are a, literally, vital aspect to insure high reliability and low maintenance cost. Adding the requirement for low emission completes the development task (Ref. 7).

A reverse flow combustor has been selected for its superior potential of:

- low temperature distribution factors
- efficient combustion with low emission
- high stability and good relight
- excellent maintainability at reduced engine length.

However, these benefits, resulting from larger volume at hand for combustion and additional transition duct length available for dilution, have to be paid for by extra effort to control wall temperatures. It has to be realized, that this is not an easy undertaking, since both wall cooling and combustion require their adequate amount of air. But today it can be reported that development progress made does safeguard a proper distribution, as verified by high pressure rig and demonstrator testing. Fig. 10 for example shows combustion chamber dome and outer transition duct after having run in the gasgenerator demonstrator. Turbine inlet temperature corresponded to Maximum Continuous rating, some 25 Centrigrade below Take-Off. Wall temperatures are very uniform and nowhere exceed 1100 K. This was made possible by first optimisation of combustion in the primary zone and second the application of efficient wall cooling technology.

In order to further reduce temperatures, or, with the same temperature level, to allow for a more costeffective wall cooling scheme, the use of thermal barrier coatings is under evalution. Test results (Fig. 11) indicate the potential to lower hot part temperatures significantly.

5. COMPRESSOR

The compressor, subject to development within the current technology demonstration, has been selected and designed by TURBOMECA to best match the requirements laid down for the baseline engine. Its performance is considered substantiated today. High design efficiency with high aerodynamic overspeed capability, low inertia rotor and extended surgemargin have, together with proper consideration of the target "into service" date, set the level to which the number of stages were reduced.

MTU's own activities, outside of the technology program mentioned before and sponsored by BMFT*, will serve to discuss the development task at hand. They are oriented at providing fundamental knowledge on axial, as well as radial compressors (Ref. 8, 9).

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* German Federal Ministery of Research and Technology

Though still research oriented in nature, an axial/radial compressor (Ref. 10) is, after having proven its potential in the rig, eventually scheduled to be demonstrator tested together with the turbine dicussed before. Fig. 12 shows a preview - the rotor at check assembly.

The further reduction of stages, i.e. augmenting stage pressure ratio without sacrifice in performance, requires techniques to be applied, which, though at MTU proven for larger massflow compressors, have to be considered new technology for this size machine (Fig. 13)

- airfoils for high supersonic flow at the inlet
- supercritical, low loss, profiles to react to high subsonic flow in the middle cascades
- boundary layer control for the extremely loaded last vane
- casing treatment in the blade tip area in addition to variable geometry vanes for improved surgemargin.

To advance technology and establish design methodology, detail analysis of individual stage performance is required. Fig. 14 for example presents the results obtained from testing the radial stage. Efficiency has been practically achieved, particularly at part load operation. Mass flow is to target.

6. FUTURE PROGRESS

While accumulating experience within the ongoing development program, the progress made and the development content have to be continuously monitored against the scenario of customer needs. Timely reaction is required, especially if priorities start drifting apart.

With the same emphasis on reduced operating cost, for a civil engine's Direct Operating Cost, lowest possible fuel consumption still outweighs complexity, but for military application, where Life Cycle Cost is concerned, reduced complexity and part cost may well become of prime significance.

A base line engine design per Fig. 15, reflects these considerations. It features a two stage axial/radial compressor driven by a single stage, highly transonic, gasgenerator-turbine. Initial and maintainance cost are greatly reduced. Because of extreme aerodynamic loading an efficiency penalty has to be accepted. This is however in part compensated - a simplified airsystem results in less leakage air and lower rotor relative temperatures permit reduced cooling airflow for the gasgenerator-turbine or an increase of stator outlet temperature. The latter effect having to be compromised for required life and improved cycle performance.

Component development has been initiated. Justifiably so, since, on the basis of the technological advances available today, it has been established that long range development potential will render superior system performance.

7. CONCLUSIONS

In conclusion then, this brief survey of development activities regarding medium power helicopter engines, was to show, that the effort concentrates on improving operating cost and reliability. Towards this end, advanced components are available today, and further progress is on route.

The coordination of three development aspects:

- improvement of analytical tools, such as design methodology and test analysis,
- optimisation and proof of performance potential in the rig, and
- substantiation of performance under engine conditions in a demonstrator,

are the key to success for the engineer.

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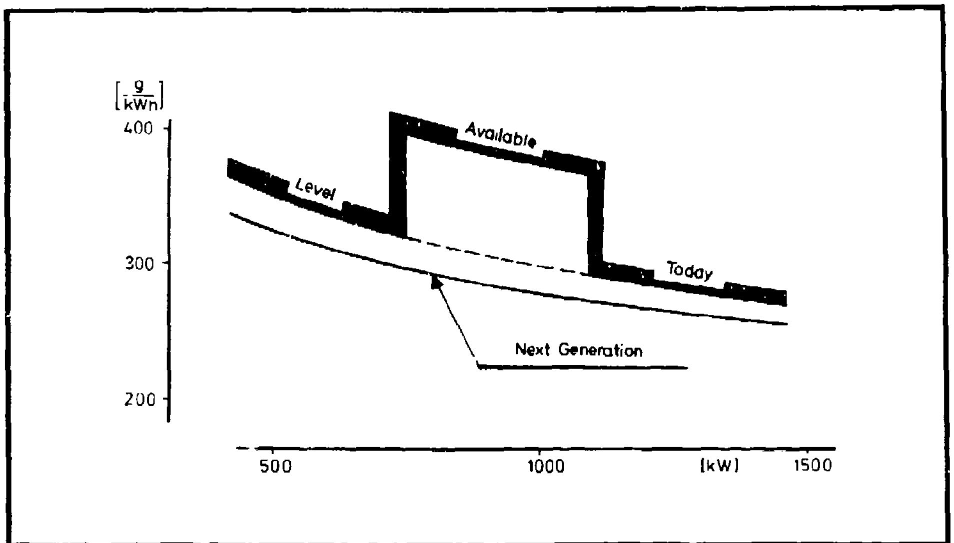

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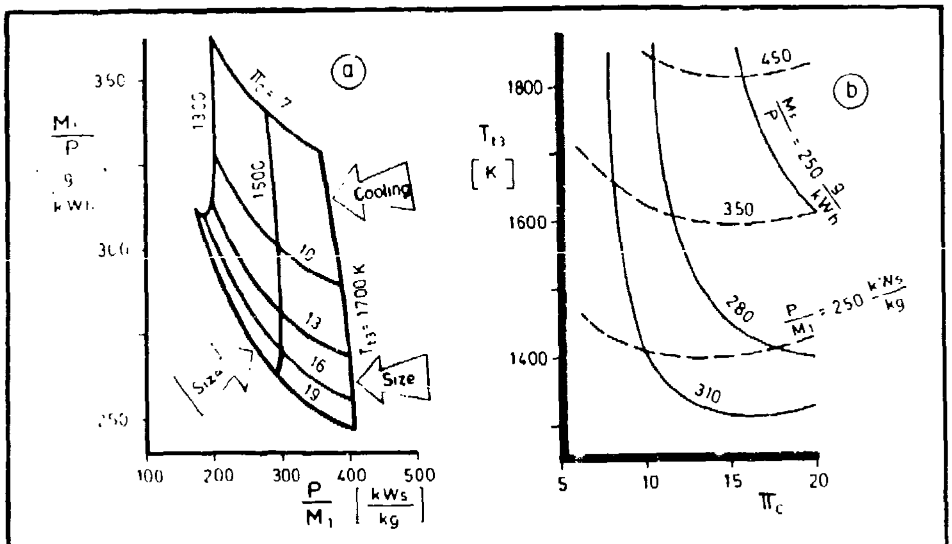
Specific Fuel Cons. ——— T.o. Power Level Available Today/Next Generation

FIGURE 1



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Thermodynamic Cyde Trend' and Limitations ('constant Losses)

FIGURE 2

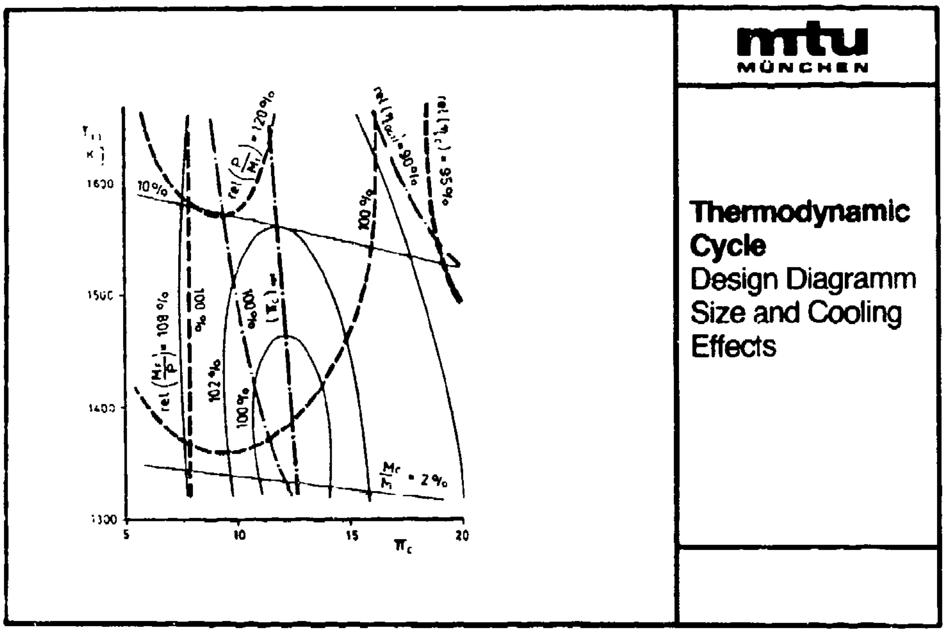

FIGURE 3

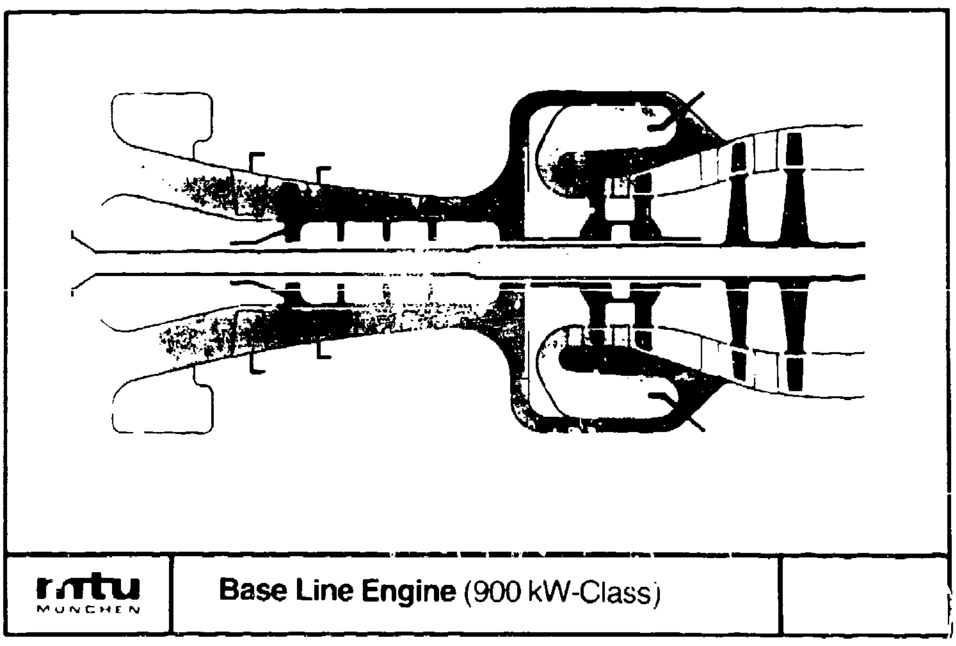

FIGURE 4

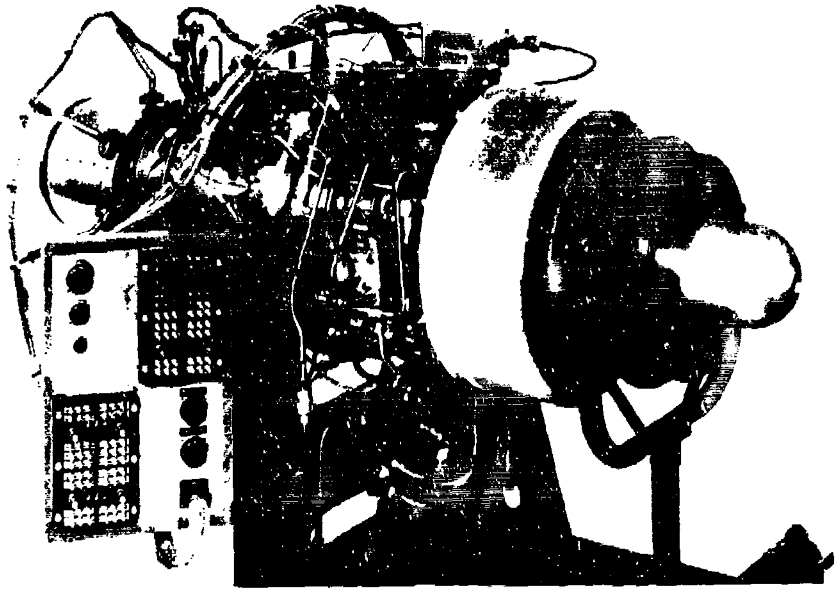

FIGURE 5

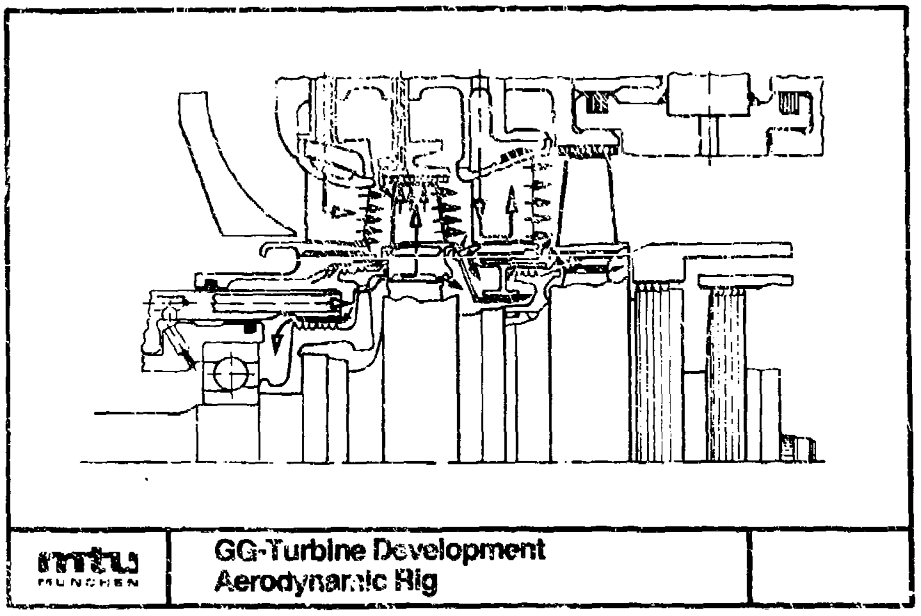

FIGURE 6

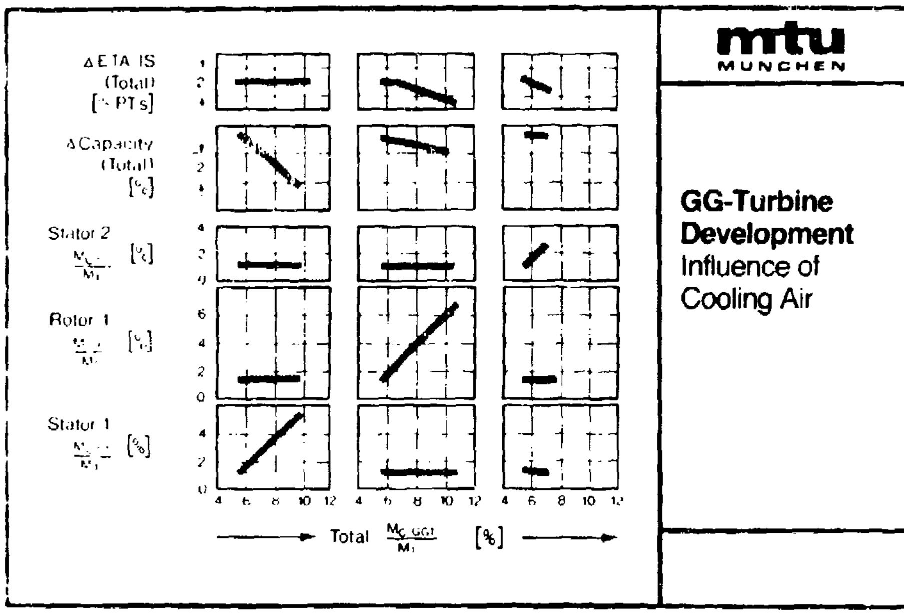

FIGURE 7

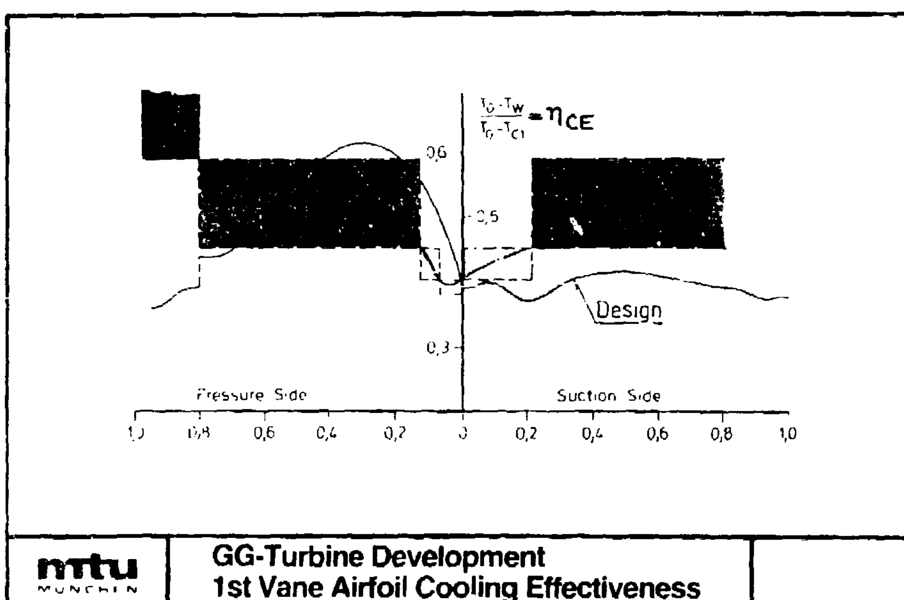

FIGURE 8

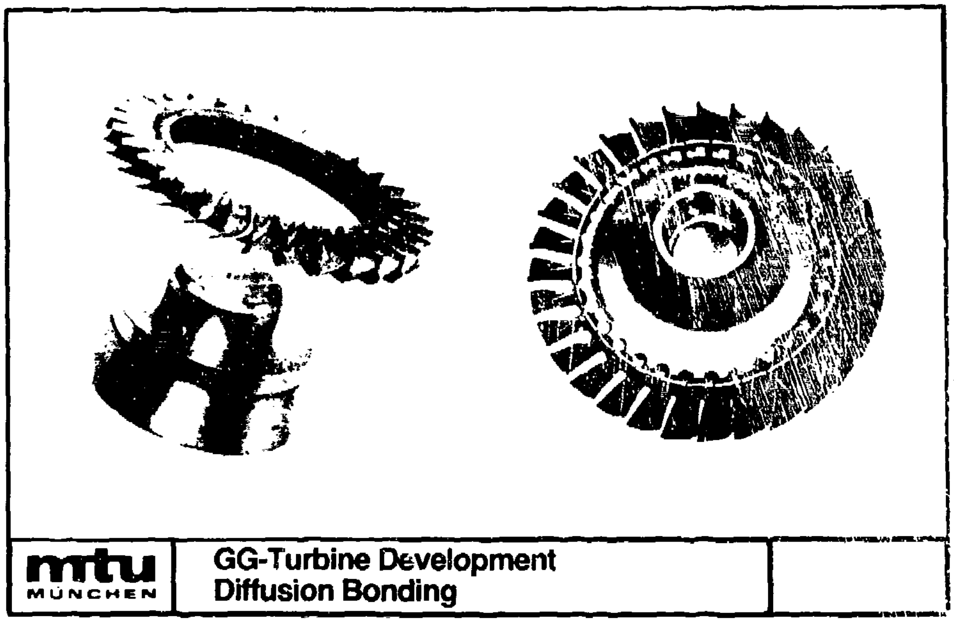

FIGURE 9

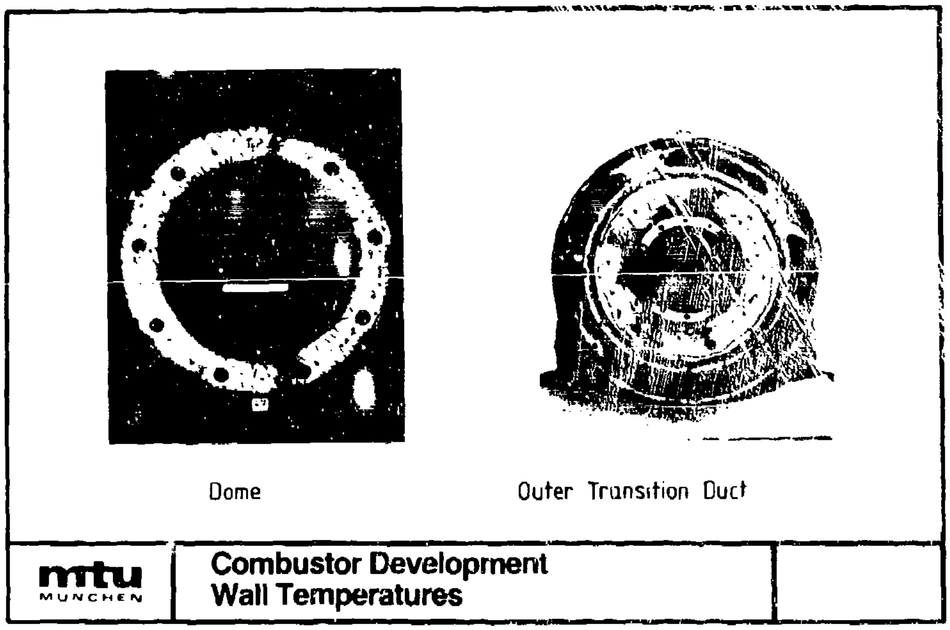

FIGURE 10

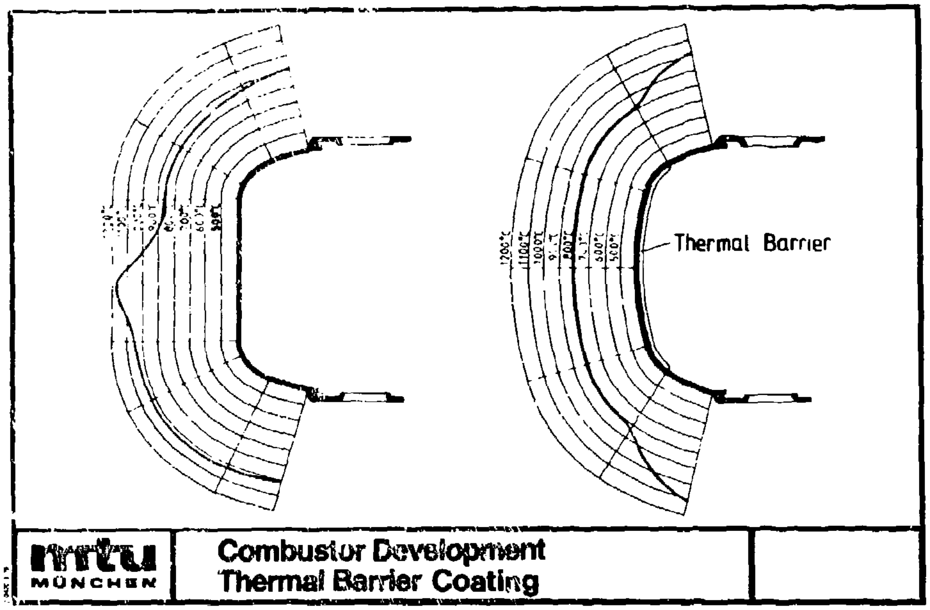

FIGURE 11

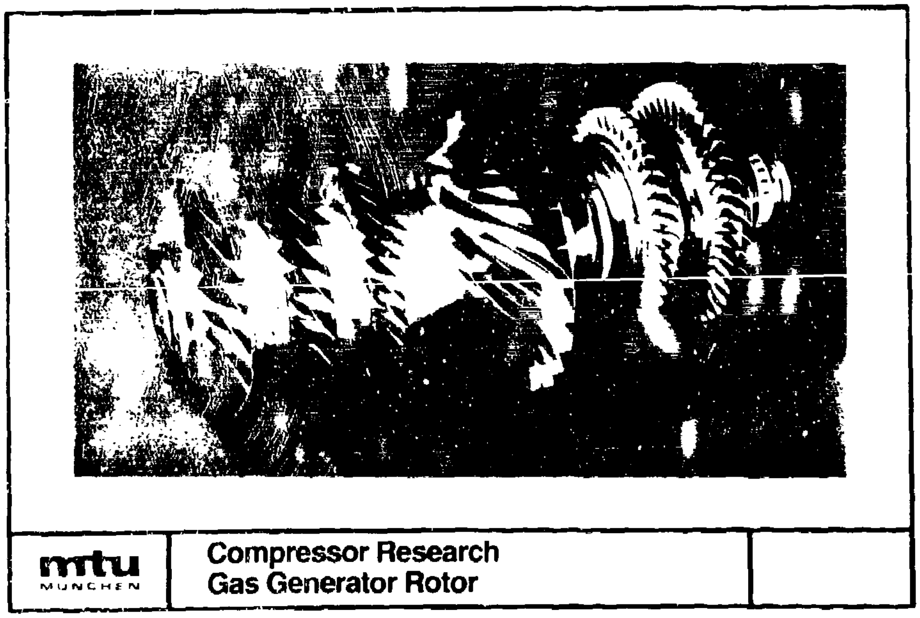

FIGURE 12

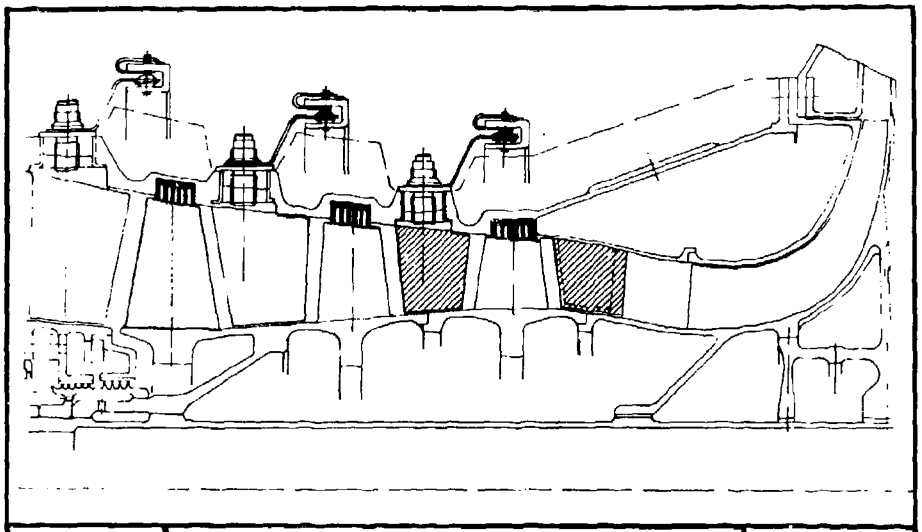

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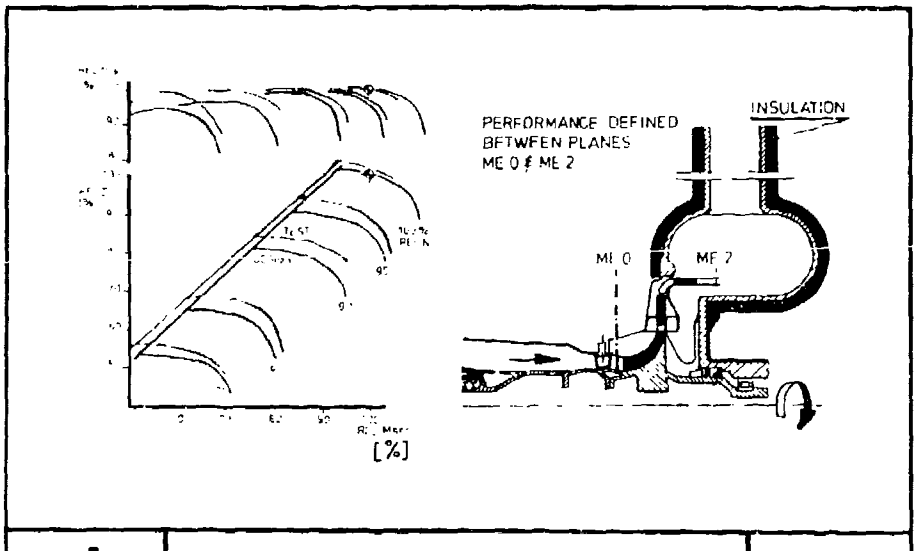
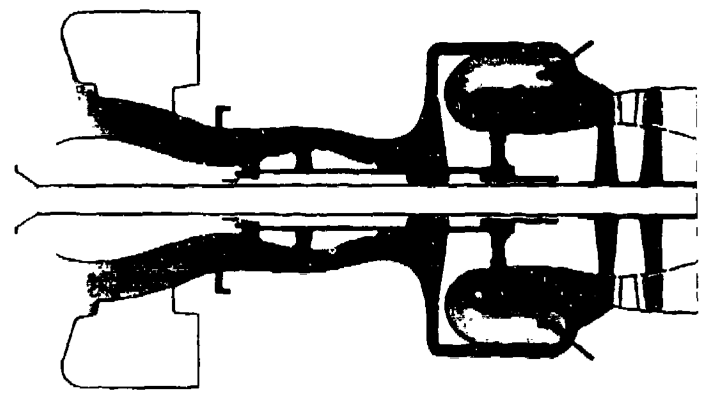
FIGURE 13



mtu MÜNCHEN | Compressor Research/Impeller Results

FIGURE 14



mtu MÜNCHEN | **Base Line Engine** (900 kW-Class) **Reaction to Change in Requirements**

FIGURE 15

DISCUSSION

K.Hart, UK

Could you comment more on your experience with thermal barrier coatings?

Author's Reply

We have applied thermal barrier coatings (zirconium oxide) to combustion chambers, blades, vanes, and platforms and have had good success, except for areas and vanes and blades directly exposed to the main gas stream.

P.Brammer, UK

Could you comment more on the subject of casing treatment?

Author's Reply

We have applied treatment to several compressors. The primary payoff is in the area of improved surge margin. However, we have yet to apply this concept to an engine that would go into production.

K.Rosen, US

Could you comment on why you selected an axial-centrifugal compressor combination?

Author's Reply

This combination gives yields of lower inertia resulting in improved acceleration characteristics.

K.Rosen, US

Could you comment on the anticipated output speed of the power turbine?

Author's Reply

It would be in the range from 28,000 to 30,000 rpm.

INFLUENCE DES NOUVELLES TECHNOLOGIES DES TURBINES SUR LEURS COMPOSANTS

par
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Bordes 64320 Bizanos
France

RESUME

La communication présentée met en évidence l'inter-dépendance de la technologie des turbo-moteurs avec celle de leurs composants.

Aprés un bref rappel des objectifs de développement des futurs moteurs d'hélicoptères de moyenne puissance, on examine les moyens de les réaliser.

Partant d'une conception classique de moteur, on montre qu'une nouvelle architecture est nécessaire et qu'elle a un fort impact sur la technologie et les performances des composants.

A leur tour, ceux-ci imposent au constructeur de nouvelles technologies de réalisation.

INTRODUCTION

La communication que nous présentons a pour objet de montrer l'interdépendance, au niveau des composants, des problèmes d'aéro-thermodynamique et des problèmes de construction. Cette interdépendance de sujets aussi divers nécessite d'être traitée par des équipes de travail pluridisciplinaires et ceci dés le stade de la conception d'un nouveau moteur.

Dans le domaine qui nous intéresse -les turbomoteurs d'hélicoptères- nous verrons que l'effet de taille est trés important et vient accroitre encore cette interdépendance de techniques trés diverses. C'est pourquoi nous limiterons notre étude à celle de turbomoteurs de taille moyenne, 400 à 1 500 kW, couvrant l'essentiel des productions de la Société TURBOMECA.

Dans le cadre qui nous est imparti nous nous sommes limités à un seul type de moteur : le turbomoteur à turbine libre afin de pouvoir montrer avec plus de détails les problèmes qui se posent au motoriste. Ceci ne peut restreindre la généralité de nos conclusions puisqu'il s'agit plutôt de décrire une méthodologie de travail que d'aboutir à la conception de telle ou telle machine particulière.

De même, les résultats de cette étude ne peuvent être considérés comme définitifs car les nombreux compromis entre des exigences contradictoires peuvent conduire à des choix différents selon l'importance plus ou moins grande que l'on voudra bien donner à tel ou tel paramètre, à telle ou telle caractéristique.

OBJECTIFS DE DEVELOPPEMENT

Pendant toute la phase de définition d'un moteur nouveau, les grands objectifs à réaliser en priorité doivent en permanence être présents dans l'esprit des concepteurs.

Ces objectifs de développement, connus de tous les motoristes, peuvent être classés en deux catégories.

La première catégorie concerne tout ce qui a trait à la diminution des coûts d'exploitation de l'hélicoptère qui utilisera le futur moteur. L'industrie des turbines est en effet maintenant en pleine phase industrielle et l'ère des pionniers, où il fallait à tout prix satisfaire le besoin technique, est dépassée. La grande majorité des constructeurs sait satisfaire ce besoin ; le problème est aujourd'hui de le satisfaire au meilleur coût pour un volume donné de production de série. Pour ce faire, les objectifs suivants sont déterminants :

- diminution du prix de revient moteur
- diminution des consommations de carburant
- diminution des coûts de maintenance
- augmentation de la fiabilité.

La deuxième catégorie d'objectifs de développement des moteurs concerne ce qui a trait à l'amélioration des performances de l'hélicoptère, soit :

- la diminution de la masse motrice
- la diminution de l'encombrement du moteur.

Ces deux paramètres influent directement sur la structure et le dessin de la cellule et contribuent ainsi à l'obtention d'un meilleur produit d'ensemble.

COMMENT REALISER CES OBJECTIFS ?

Nous commencerons par les examiner un à un.

Le prix de revient du moteur est de toute évidence un paramètre très important. Nous laisserons de côté le facteur de décroissance du prix en fonction du rang de série (loi de Wright) parce que, de par sa généralité, il n'est pas influencé par la conception du moteur et la qualité de ses composants. Ce paramètre peut être facilement éliminé en raisonnant pour un rang de moteur fixé dans la série : 100e, 1 000e, etc... Dés lors, la diminution du coût du moteur passe par celle du nombre de ses pièces constitutives. Ce nombre de pièces est à son tour fonction de la conception et du nombre des composants, ces deux éléments s'influençant mutuellement. Par conception, nous entendons l'architecture générale du moteur, terme que nous conserverons dans la suite de notre exposé. C'est l'architecture du moteur qui définit le nombre et la disposition des arbres, des roulements, des paliers et des composants. Pour les composants - (étages de compression et de détente) - plus leur nombre sera faible, plus sera faible le nombre des aubages fixes et mobiles et plus sera faible le prix de revient moteur. En définitive, nous retiendrons comme fondamental pour le coût du moteur son architecture et le nombre des composants.

Autre objectif à atteindre : une faible consommation spécifique de carburant. Ici, c'est le cycle thermodynamique du moteur qui est impliqué, c'est à dire essentiellement trois facteurs :

- le taux de compression
- la température entrée turbine
- les rendements des composants.

L'historique des turbomachines a bien montré, à ce point de vue, une évolution plus ou moins continue de ces trois paramètres mais qui a conduit au moins pour les deux premiers à une augmentation de la complexité des machines : nombre des composants et architecture, donc en conflit direct avec l'objectif "prix de revient". Il existe cependant une voie pour résoudre ce problème, c'est l'utilisation des grandes vitesses périphériques. Il sera donc possible d'augmenter, dans les étages élémentaires, taux de compression et de détente et ainsi d'améliorer le cycle thermodynamique sans compliquer l'architecture du moteur. Cette voie est celle suivie par TURBOMECA. Ce n'est pas la plus facile mais il est certain qu'elle est trés prometteuse. Elle n'est possible que si l'on maitrise bien l'aérodynamique des écoulements transsoniques et supersoniques sinon les rendements des composants vont se dégrader et annuler le bénéfice que l'on escompte.

L'objectif d'abaissement des coûts de maintenance est obtenu par la conception modulaire des moteurs, conception pratiquée actuellement par la plupart des constructeurs. Là encore, c'est l'architecture du moteur qui permettra de réaliser cette exigence.

L'objectif de fiabilité sera atteint par une construction "robuste" des différents éléments du moteur. Les études et les efforts doivent porter sur les durées de vie, les vies cycliques, la protection contre l'absorption des corps étrangers, les rétentions des pièces tournantes, les effets des pollutions diverses (eau, glace, sable, etc...) et l'environnement moteur (distorsions de flux, reingestions de gaz brûlés, etc...).

L'objectif de diminution de la masse motrice peut, sur certains points, être contradictoire avec l'objectif précédent. Pour sa réalisation, il pourrait nécessiter en effet une construction légère et trés élaborée au niveau des pièces élémentaires qui va s'opposer à l'aspect "robustesse" indiqué ci-dessus. Il y a donc, là encore, un difficile compromis à obtenir. Par contre, le meilleur moyen de réaliser une masse spécifique faible réside bien dans une conception simple du moteur, c'est à dire que l'on retrouve encore le paramètre "architecture".

Le dernier objectif à atteindre, minimiser l'encombrement, sera lui-aussi grandement fonction de l'architecture, mais également du nombre des composants, leur diminution étant évidemment un facteur à rechercher.

Nous pouvons maintenant faire la synthèse des différents moyens retenus pour réaliser les objectifs que nous avons précédemment définis. Trois facteurs essentiels subsistent et de la manière dont ils pourront être traités dépendra pour beaucoup la réussite du moteur ainsi conçu. Pour nous, le "triplé gagnant" est bien le suivant :

Architecture - cycle- nombre de composants. L'approche traditionnelle pour traiter ce triplet comportait une certaine hiérarchie, ou pour le moins, une certaine chronologie :

1. Détermination du cycle.
2. Nombre des composants.
3. Architecture moteur.

L'approche actuelle est fondamentalement différente et consiste à traiter simultanément les trois éléments du triplet : toute option sur l'un de ces éléments devra être immédiatement examinée au niveau de ses implications sur les deux autres, ce qui impose au motoriste une communication et un va-et-vient permanent de l'information au sein des différentes équipes de travail.

DESCRIPTION DES TURBOMOTEURS DE CONCEPTION CLASSIQUE

Dans la gamme des puissances moyennes, les derniers turbomoteurs TURBOMECA de "conception classique" qui ont été lancés en série, l'ARRIEL (figure 1) et le MAKILA (figure 2) sont caractérisés par :

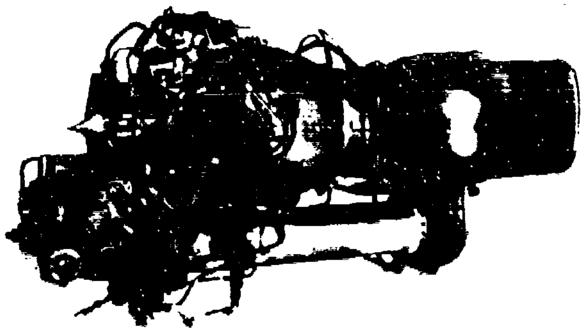


Figure 1

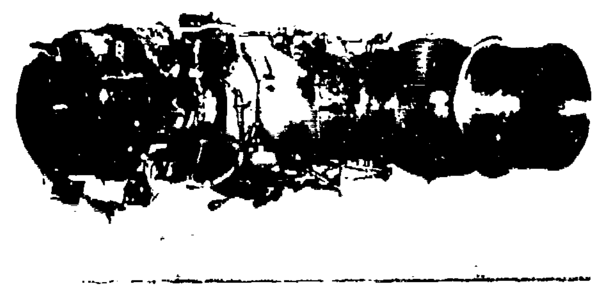


Figure 2

. <u>Un générateur de gaz composé</u> :

- d'un compresseur comprenant un étage centrifuge couplé selon le cas à 1 ou 2 ou même 3 étages axiaux, l'ensemble réalisant un taux de compression π = 8 à 10.
- d'une turbine, à deux étages dont la température d'entrée (TET) limitée à 1 000°, 1 050°C évite d'utiliser un système de refroidissement (toujours couteux et délicat dans les petites turbomachines). La turbine et le compresseur constituent un simple corps utilisant deux arbres (figure 3) ou un arbre sur trois paliers (figure 4)

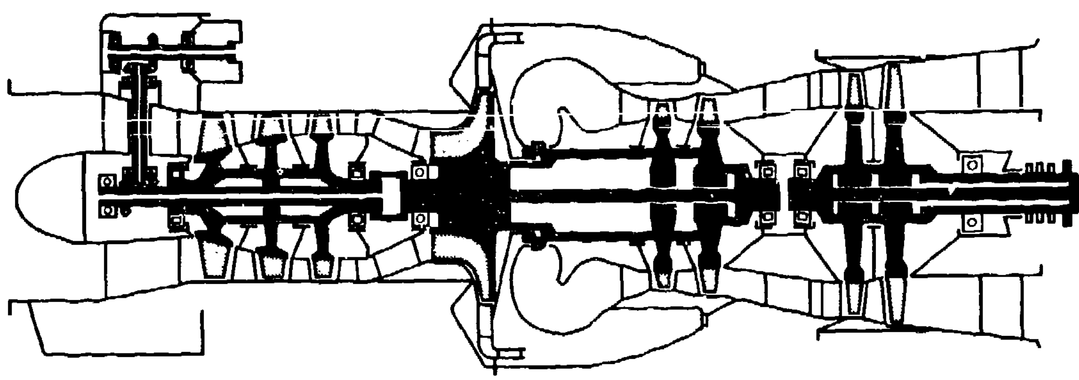


Figure 3

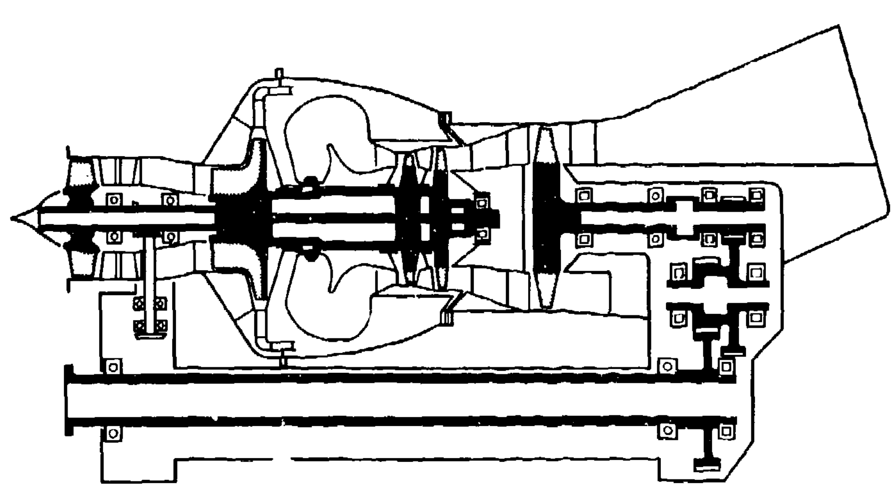


Figure 4

- d'une chambre de combustion placée entre compresseur et turbine. Elle est à injection centrifuge de carburant au moyen d'une roue alimentée latéralement, (dispositif plus simple s'il est comparé à l'alimentation par tube central réalisée sur les moteurs de la génération précédente : TURMO - ASTAZOU).

. Une turbine de puissance à un ou deux étages dont la puissance est fournie à la boite de transmission principale de l'hélicoptère soit en prise directe (figure 3), soit par l'intermédiaire d'un réducteur (figure 4). Les différents types de prise de mouvement peuvent imposer la réalisation d'un arbre de puissance parallèle au corps du générateur (figure 4).

Ces choix conduisent à une architecture moteur dans laquelle le nombre de paliers varie de 5 à 7 et le nombre d'enceintes de paliers de 3 à 4, imposant ainsi autant de dispositifs de servitude (alimentation et récupération d'huile, mises à l'air, étanchéité).

CONCEPTS D'AMELIORATION DES TURBOMOTEURS

La nouvelle architecture qui s'impose conduit à repenser successivement les ensembles tournants, la chambre de combustion, le compresseur, la turbine.

Ensembles tournants : L'objectif de simplification conduit à réaliser un générateur de gaz comprenant un seul arbre supporté par deux paliers. La turbine de puissance aura une disposition semblable. Cependant, afin de concilier encombrement et poids minimums avec une prise de puissance à l'avant ou à l'arrière, l'arbre de puissance sera concentrique à celui du générateur. Cette disposition réduit le nombre d'enceintes de paliers et permet aisément le développement en turbopropulseur et turbofan ce qui pourra réduire le prix de revient du générateur de gaz.

Chambre de combustion : L'architecture définie précédemment implique de raccourcir au maximum les distances entre paliers à la fois pour le générateur et l'arbre de puissance afin de minimiser les problèmes de vitesses critiques et de vibrations. Pour cela, la chambre de combustion TURBOMECA classique, à injection centrifuge, est abandonnée au profit d'une chambre à flux inversé (figures 10 et 11). Le volume de chambre (fonction du débit d'air et du taux de compression) se trouve transféré au-dessus de la turbine HP et la distance entre compresseur et turbine est réduite à la simple implantation du coude d'amenée des gaz à l'entrée du distributeur 1er étage. Il n'y a pratiquement pas d'augmentation de maître-couple du moteur car ce type de chambre s'intègre parfaitement derrière un étage de compression centrifuge. Ce choix est expliqué ci-après.

Compresseur : Chercher à diminuer le nombre de composants conduit à adopter un seul étage centrifuge à vitesse périphérique élevée et bon rendement. Ce dernier peut faire le même travail de compression que 4 à 6 étages axiaux transsoniques. Ce type de compresseur a des aubages moins fragiles qu'une roue axiale et de ce fait présente un avantage du point de vue fiabilité. Ce choix s'impose donc, à condition d'obtenir un étage centrifuge à hautes performances : vitesse élevée et bons rendements. Or, avec les progrés effectués en ce domaine depuis quelques années on sait maintenant concevoir et réaliser de tels centrifuges avec des rendements polytropiques au moins égaux à ceux des étages axiaux qu'ils remplacent.

Turbine haute pression : Le but à rechercher est le même que pour le compresseur. Compte-tenu du coût de cet organe et de sa difficulté de mise au point, on cherchera à réaliser la détente en un seul étage et on reculera le plus possible l'introduction du refroidissement des pales mobiles.

Par la considération du couple : architecture-composants, on aboutit ainsi au générateur de gaz le plus simple et le plus compact qui soit (figure 5), comprenant :

. une chambre de combustion à flux inversé
. un compresseur centrifuge à hautes performances
. une turbine HP monoétage non refroidie
. un arbre supporté par deux paliers seulement.

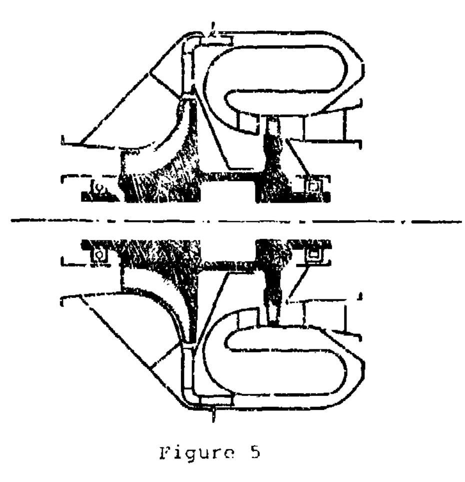


Figure 5

Cycle thermodynamique : L'étage centrifuge est-il suffisant pour la compression ? L'examen du cyle thermodynamique, troisième terme du triplet mis en relief précédemment, va nous donner la réponse.

En effet, avec un étage centrifuge dont le taux de compression au régime nominal serait de 7 à 9, associé à une turbine HP fonctionnant dans la gamme des températures 950/1 050°C, les calculs de cycle montrent que les consommations de carburant seraient situées dans la fourchette de 330 à 350 g/kW.h. Avec les augmentations successives des prix pétroliers, ces valeurs qui auraient été acceptables il y a encore quelques années ne le sont plus aujourd'hui, pour un moteur nouveau. Nous sommes donc obligés de compliquer le compresseur pour aboutir à un moteur de meilleure qualité.

La vraie question est donc de savoir jusqu'à quel point il est nécessaire de compliquer ce moteur. La réponse n'est pas simple, mais surtout elle n'est pas unique. La seule considération des courbes classiques des puissances et consommations spécifiques (figure 6) en fonction des paramètres majeurs : taux de compression et T.E.T. (températures entrée turbines) est tout à fait insuffisante. Leur seul intérêt est de montrer que la T.E.T. doit suivre l'augmentation du taux de compression ; mais il serait dangereux d'utiliser les valeurs qui semblent optimales.

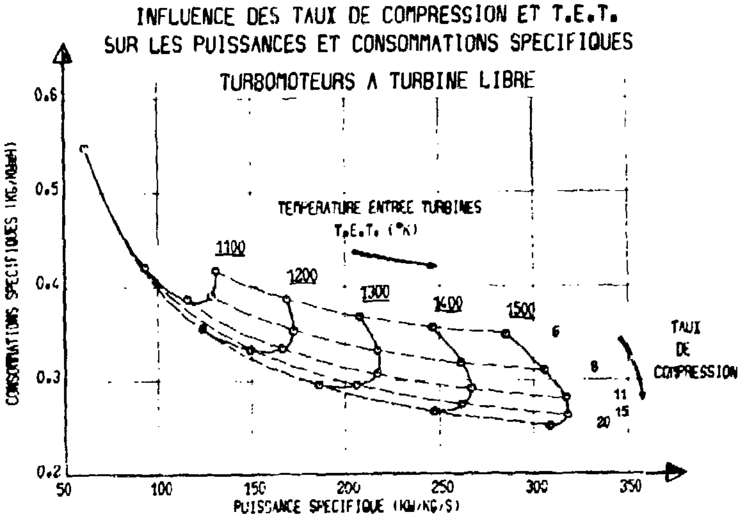


Figure 6

En effet, notre problème est bien de construire un moteur de puissance nominale donnée. Or, plus le cycle est poussé, plus les débits d'air, donc les dimensions du moteur sont réduites : d'où les pertes accrues par écoulements secondaires venant amplifier celles dues aux faibles valeurs des nombres de Reynolds. Ceci entraine des dégradations des rendements des composants venant à l'encontre des bénéfices escomptés par les améliorations des cycles thermodynamiques. Cet effet est de première importance pour les machines de faibles et moyennes puissances dont nous parlons et c'est ce qui les différencie fondamentalement des grosses turbomachines de l'aviation civile ou militaire. Pour illuster ces effets, nous prendrons l'exemple concret d'un moteur de 600 kW au régime nominal et nous comparerons les résultats de trois projets complets effectués avec des cycles et des architectures de complexités croissantes.

TURBO-MOTEURS DE 600 KW

	MOTEUR N° 1	MOTEUR N° 2	MOTEUR N° 3
TAUX DE COMPRESSION	8	11	14
TEMPERATURES ENTREE TURBINES (°K)	1323	1386	1450
DEBITS D'AIR (KG/S)	2.59	2.33	2.32
FORMULE COMPRESSEUR	1C	2A+1C	3A+1C
FORMULE TURBINES	1TG/1TL	1TG/1TL	2TG/2TL
REFROIDISSEMENT TURBINES	NON	NON	OUI

Figure 7

La figure n° 7 donne la description des cycles et composants de ces moteurs. Le moteur n° 2 diffère du premier principalement par son compresseur et sa T.E.T. un peu plus élevée. Par contre, le moteur n° 3 est nettement plus compliqué surtout dans la partie détente : 4 étages de turbine dont 2 ont des pales refroidies.

La figure suivante (figure 8) met en relief les effets des cycles étudiés, sur les dimensions des composants les plus critiques ainsi que les rendements qui en découlent. Les performances globales sont aussi indiquées.

Le résultat le plus important est que la complication du moteur n° 3 n'est pas justifiable au vu des résultats : gains de l'ordre de 1 % seulement sur consommations et puissances spécifiques pour un accroissement sensible du nombre des composants majeurs sans parler du refroidissement des pales de turbine du générateur. Au contraire, l'adjonction de deux étages axiaux au compresseur centrifuge permet un gain de l'ordre de 10 % sur les performances du moteur n° 1, cette configuration est donc payante. Bien sûr, le choix de la puissance nominale du moteur est fondamental. Si au lieu de 600 kW nous avions choisi 1 400/1 500 kW il est fort probable que le moteur n° 3 serait proche de la solution optimale.

TURBO-MOTEURS DE 600 KW

	MOTEUR N° 1	MOTEUR N° 2	MOTEUR N° 3
HAUTEUR DE PALES SORTIE CENTRIFUGE (MM)	7.2	6.2	5.1
HAUTEUR DE PALES TURBINE GENERATEUR (MM)	17.5	14.5	10.9
DEBIT D'AIR PERDU POUR LA DETENTE (%)	1.5	1.5	4.5
RENDEMENTS POLYTROPIQUES : (%)			
. COMPRESSEUR	83.9	83.9	83.3
. TURBINE GENERATEUR	83.9	82.6	80.9
. TURBINE DE PUISSANCE	86.0	85.6	86.1
CONSOMMATIONS SPECIFIQUES (G/KW.H)	327	299	296
PUISSANCES SPECIFIQUES (KW/KG/S)	227	252	254

Figure 8

De cette brève étude, nous conclurons que pour un niveau donné de la technologie des composants :

- l'optimum du taux de compression est fortement dépendant de la puissance nominale du moteur
- cet optimum se répercute directement et de façon importante sur le nombre des composants et l'architecture du moteur.

Compte-tenu des remarques relatives aux ensembles tournants, l'adjonction d'une turbine libre sur ce générateur de gaz conduit à une conception de moteur que l'on peut comparer aux générations actuelles (figure 9).

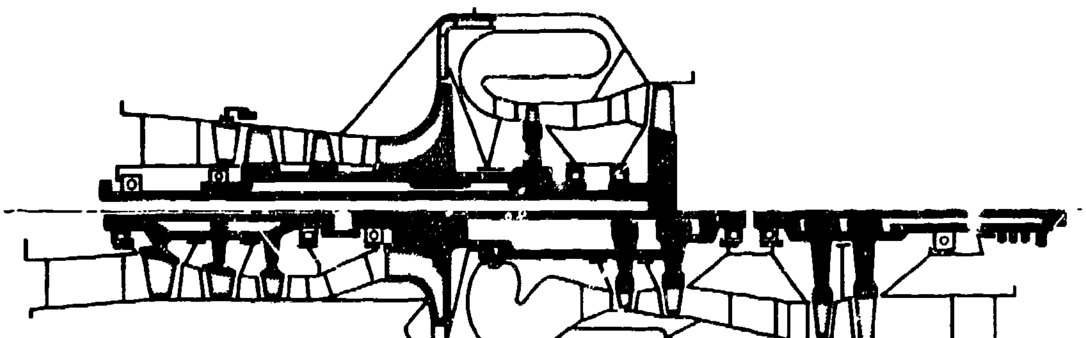


Figure 9

TECHNOLOGIE DES COMPOSANTS

Pour chaque composant, il est nécessaire d'examiner les raisons du choix effectué pour réaliser le compromis architecture-cycle et surtout d'analyser les répercussions sur leur technologie propre et sur les problèmes soulevés par leur assemblage et leur interdépendance au sein du moteur.

Chambre de combustion à flux inversé :

Les raisons d'architecture moteur ont déterminé ce choix, et les problèmes essentiels suivants seront à résoudre.

Sous l'aspect performances :

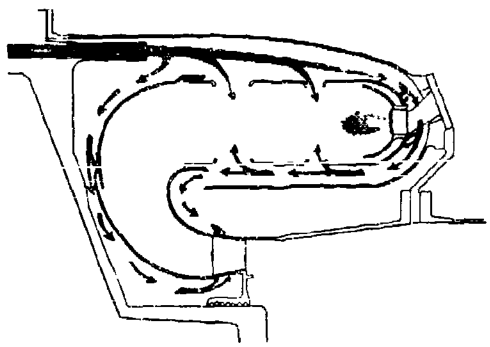
- La recherche de bons rendement, y compris à bas régime, se fera par une étude de l'aérodynamique de la zone primaire de combustion, la perte de charge demeurant cependant plus élevée que dans une chambre à injection centrifuge.

- La pollution même avec l'utilisation de cannes à prévaporisation restera aussi légèrement supérieure aux résultats d'une chambre conventionnelle TURBOMECA.

Sur le plan mécanique :

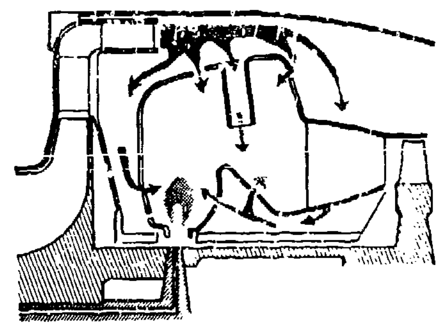
- Il est nécessaire d'obtenir une homogénéité circonférentielle et un profil radial de températures acceptables. Notons que ce dernier présentera un profil "bombé" plus accentué à cause de l'augmentation du débit d'air de refroidissement des parois. La chambre à flux inversé présente en effet, à même charge aérodynamique et donc à volume identique, plus de surface à refroidir que le modèle classique TURBOMECA.

- Cependant, cette réalisation si elle nécessite des systèmes d'injection à pression plus élevée que ceux requis pour l'injection centrifuge permet de s'affranchir des problèmes de tenue mécanique qui peuvent être rencontrés sur les roues d'injection.



CHAMBRE DE COMBUSTION A FLUX INVERSE

Figure 10



CHAMBRE DE COMBUSTION A INJECTION CENTRIFUGE

Figure 11

Turbine haute pression :

Une solution monoétage à la limite du refroidissement a été retenue pour satisfaire le compromis coût-performance. Ce choix est possible à condition d'augmenter la charge aérodynamique ce qui aura pour conséquence de diminuer la température totale relative à l'entrée de la roue mobile. Une détente au moyen de 2 étages ne réalisera pas cette condition et pour une même T.E.T. le 1er étage sera nécessairement refroidi.

Ce choix permet donc d'éviter le refroidissement et les problèmes qui lui sont associés dans le cas de petites turbomachines :

- pertes aérodynamiques (réinjection d'air) et thermodynamiques (prélèvement d'air de refroidissement)

- difficulté de réalisation des circuits de refroidissement dans des pales de faible dimension, d'où une augmentation des coûts.

Il est donc judicieux de n'avoir à traiter ces problèmes que sur un seul étage lors d'un développement qui viserait à augmenter la puissance spécifique de ce moteur. Cela ne sera possible que si le niveau de température choisi compense largement les chutes de rendement associées à l'introduction du refroidissement.

Dans le choix de base effectué, "monoétage non refroidi", les problèmes suivants doivent être maitrisés :

Aérodynamique :

- Assurer des déviations fluides importantes tout en minimisant les Mach d'écoulements locaux afin d'éviter tout décollement prématuré de la couche limite.

- Réduire les pertes par jeu en sommet de pale ; la faible hauteur de ces dernières en accentue l'importance (à une augmentation de 1 % du jeu relatif correspond une chute de rendement de l'ordre de 1,5 point). Le refroidissement des enveloppes de turbine est donc indispensable et des barrières thermiques peuvent lui être associées afin de contrôler un jeu minimum dans toutes les conditions de fonctionnement.

- Une éventuelle giration résiduelle de l'écoulement en sortie de turbine et son amplification possible dans le canal inter-turbines doit être acceptée au niveau des bras supports du palier arrière.

Notons que l'adoption d'un système d'arbres contrarotatifs permet d'utiliser cette giration pour diminuer la charge aérodynamique du distributeur de turbine libre.

Mécanique :

L'augmentation des vitesses périphériques et la nécessité de conserver un trou central ont pour conséquence d'augmenter les contraintes, dans le disque de turbine, en pied de pales, et les pressions spécifiques sur les brochages. Des compromis entre hauteur d'aube, longueur de corde, nombre d'aubes permettent d'aboutir à un dimensionnement convenable.

Les réalisations de turbines "monobloc" coulées, sur des moteurs type ARRIEL sont remplacées par des constructions disques forgés, pales coulées en choisissant un matériau propre à chaque fonction :

- disques : tenue à la fatigue oligocyclique au moyeu et fatigue thermique à la jante
- pales : fluage et corrosion hautes températures.

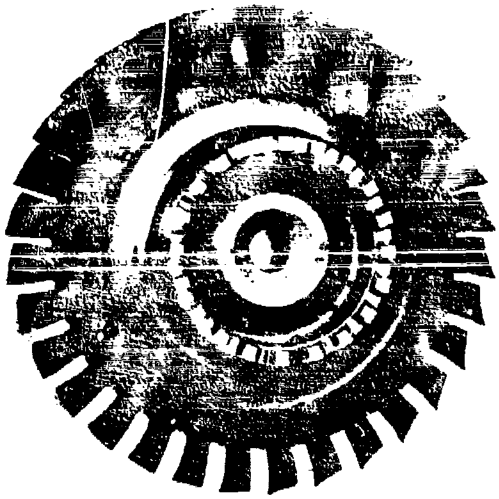


Figure 12

COMPRESSEUR

Nous avons montré qu'un seul étage centrifuge était en général insuffisant pour assurer l'ensemble de la compression. Pour compléter cette dernière, deux configurations sont possibles, soit adjoindre des étages axiaux, soit utiliser un autre étage centrifuge.

Cette deuxième solution conduit à un dessin compliqué et ne peut se justifier que pour un taux de pression global élevé (15 à 16). Dans ces conditions, il faudra 2 étages de turbine pour détendre les gaz et des T.E.T. élevées nécessitant le refroidissement de la partie haute température, des turbines. On voit donc que ce choix ne peut être retenu que pour des moteurs de puissance relativement élevée, en général au-delà de 1 500 kW.

L'autre solution, l'adjonction d'étages axiaux, est donc celle qu'il nous faut adopter : mais là encore en cherchant à en minimiser le nombre. On est donc conduit à utiliser des étages transsoniques fortement chargés. Enfin, pour des raisons évidentes de compacité de l'ensemble on s'efforcera de coupler directement l'axial au centrifuge sans col de cygne intermédiaire.

La configuration de base, maintenant bien définie, examinons pour chaque organe les problèmes technologiques à maitriser.

COMPRESSEUR CENTRIFUGE

Sur le plan aérodynamique, l'adoption des grandes vitesses phériphériques et des taux de compression élevés qui en découlement impliquent pour obtenir de bons rendements :

. l'utilisation de concepts à degré de réaction élevé

. la connaissance précise des caractéristiques de l'écoulement fluide-vitesses, pressions, températures- en tout point des rotors et diffuseurs et ceci pour des régimes mixtes : subsoniques - supersoniques.

Les nouveaux codes de calcul disponibles ainsi que les expérimentations intensives sur bancs séparés ont abouti à une véritable mutation des formes en ce domaine. Les figures 13 et 14 le montrent clairement.

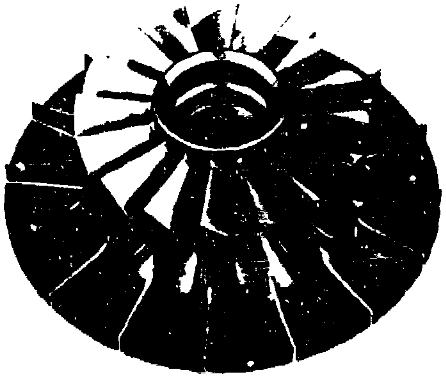


Figure 13

Figure 14

Sur le plan mécanique, les problèmes ne sont pas moindres :

. Vitesses périphériques augmentées par rapport à l'ancienne génération.

. Températures plus élevées dues à la forte compression globale.

. Trou central important nécessaire au passage des arbres.

Là encore les méthodes modernes de calcul des structures et l'utilisation des meilleurs matériaux disponibles permettent d'aboutir à un dimensionnement capable d'assurer en utilisation une longévité et une fiabilité satisfaisantes.

COMPRESSEUR AXIAL

Le couplage serré des compresseurs axiaux au centrifuge diminue nécessairement la vitesse périphérique des derniers étages axiaux et rend ainsi plus difficile l'obtention des taux de compression unitaires élevés. Par ailleurs, le passage des arbres sous le compresseur axial a l'effet contraire d'augmenter excessivement les diamètres de la veine d'air et par voie de conséquence les nombres de Mach relatifs sur les mobiles: d'où des difficultés accrues pour assurer de bons rendements.

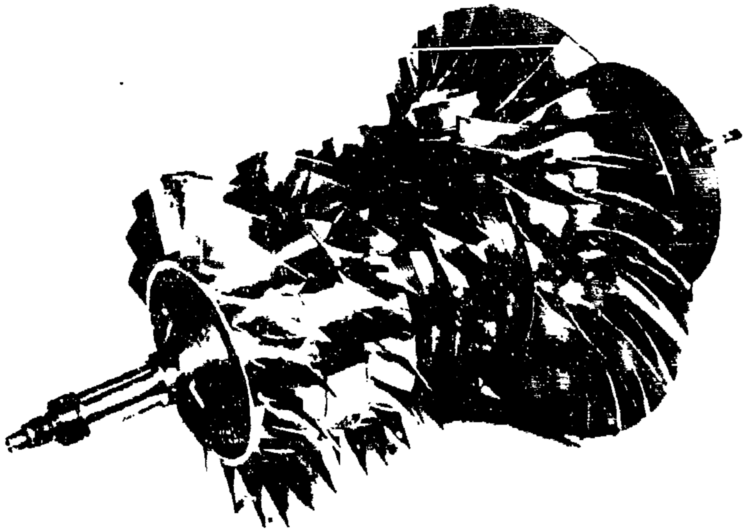


Figure 15

Enfin, pour améliorer les performances en régime partiel et obtenir un maniement correct du moteur, ces compresseurs à forts taux de compression globale doivent avoir une certaine flexibilité. La solution à 3 arbres concentriques nous parait trop complexe donc trop chère pour être retenue, au moins pour notre gamme de puissances ; nous lui préférerons donc l'utilisation de grilles distributrices à calage variable.

Turbine de puissance :

Une construction monoétage peut être généralement adoptée pour des moteurs de puissance inférieure à 700 kW. Mais cela n'est pas une règle. En fait, le choix du nombre d'étages de turbine libre dépend à la fois, du canal inter turbines (forme - pertes), de la vitesse de rotation, du rendement attendu. Ces paramètres sont bien sûr interdépendants. Enfin, la vitesse de rotation est souvent un compromis réalisé entre les performances et les exigences de l'installation sur hélicoptère : prise directe ou sortie au travers d'un réducteur intégré au moteur.

Arbres et rotors :

L'arbre de puissance concentrique au rotor du générateur de gaz a des vitesses critiques disposées en dehors des régimes de fonctionnement de la turbine libre. Pour cela, il faut connaitre parfaitement les régimes de survitesses (dérives de régulation, etc...) rencontrés sur hélicoptère en utilisation.

Deux types de solutions sont envisageables :

- Une solution "sous-critique" (1re vitesse critique d'arbre > vitesse de fonctionnement) nécessite un couple diamètre-longueur d'arbre toujours difficile à réaliser pour un moteur de la taille envisagée. Ce choix peut conduire à la réalisation d'un palier inter-arbres (solution technique plus onéreuse et délicate dans le cas d'arbres contrarotatifs).

- Une solution "surcritique" (1re vitesse critique d'arbre < vitesse de fonctionnement) qui impose un régime de rotation situé entre la 1re et la 2e vitesse critique d'arbre avec des marges de fonctionnement suffisantes.

  Dans ce cas, afin de minimiser les déformations de l'arbre lors du franchissement de la 1re vitesse critique, on la placera sur un régime de fonctionnement relativement bas. Cela est possible en travaillant sur la souplesse des paliers, l'équilibrage et la géométrie des arbres.

Paliers :

Avec des vitesses élevées et la réalisation de l'arbre traversant, les roulements sont soumis à des conditions de fonctionnement plus critiques :

- N x Dm - (vitesse x diamètre moyen) - plus élevés (Figure 16)
- ambiances de températures plus sévères sur le palier arrière.

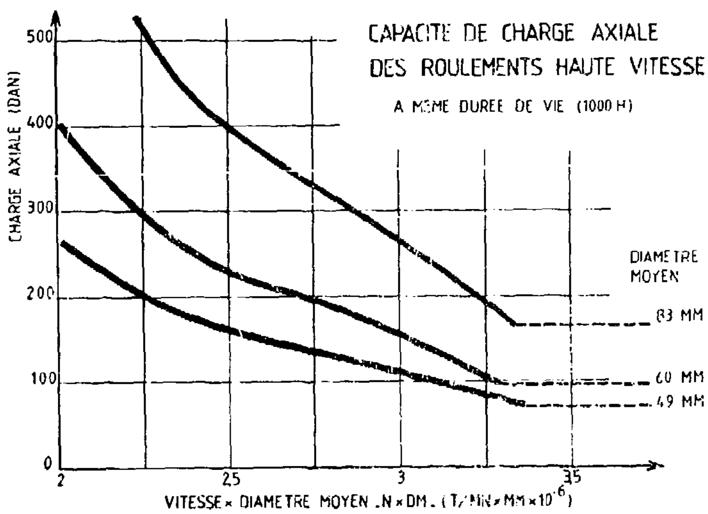


Figure 16

Ainsi, sur les paliers du générateur de gaz, l'augmentation de force centrifuge sur les éléments roulants est associée à une poussée axiale généralement élevée. Il en résulte un accroissement de charge sur les bagues extérieures accompagné d'échauffements sur les bagues intérieures

Pour obtenir les durées de vie demandées :

- les matériaux "acier rapide" doivent être adoptés pour les pistes et les éléments roulants

- la lubrification de la bague intérieure est particulièrement étudiée
- le comportement de la cage est analysé afin de permettre des fonctionnements lors des réductions ou des absences momentanées de lubrification.

Les garnitures mécaniques conventionnelles (joints type Sealol) ne supportant pas des vitesses supérieures à 70 m/s, les étanchéités de paliers doivent être à labyrinthes. L'encombrement radial est plus faible et ils ne nécessitent aucune servitude (absence de lubrification). Les consommations d'air sont minimisées par la diminution des jeux réalisée par la mise en place de nouveaux matériaux abradables. Ces revêtements développés pour des températures pouvant dépasser 800 °C sont aussi appliqués sur les étanchéités inter étages de compresseur et de turbine afin d'améliorer les performances de l'ensemble.

CONCLUSIONS

Au cours de notre étude, il est apparu fondamental, lors de la conception de nouveaux moteurs d'hélicoptères, de traiter simultanément les problèmes de cycles thermodynamiques, de nombre de composants et d'architecture moteur.

Un résultat ne peut être atteint que par le développement parfois difficile de composants à hautes performances s'intégrant parfaitement dans l'architecture préalablement définie. Ces compromis permettront de réaliser des solutions attrayantes au point de vue prix, consommation, fiabilité.

En outre, l'interdépendance étroite de l'architecture du moteur et de la technologie des composants, implique une coopération active et simultanée des équipes de travail participant à la conception et à l'étude de toute machine nouvelle.

Plutôt que les conclusions auxquelles nous sommes parvenus sur tel projet de moteur bien particulier, conclusions toujours fragiles car dépendant des progrès techniques et des données économiques du moment, on retiendra surtout la méthodologie de conception adoptée.

Ce n'est qu'en la respectant fidèlement, tel est du moins notre avis, que l'on pourra mener à bien la réalisation d'un nouveau produit techniquement réussi.

DISCUSSION

D.Hennecke, Ge

In Figure 10 you show a reverse flow combustor with an injection system that appears to be an air blast vaporizer. Could you comment on your experience with this kind of system and whether or not it has advantages over the vaporization system?

Author's Reply

This particular figure is only schematic. From our experience, we have concluded that the vaporization system is the best and that air assist is not necessary for these small power applications.

AERODYNAMIC COMPONENTS FOR SMALL TURBOSHAFT ENGINES

by

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SUMMARY

Future developments of advanced helicopter engines are projected from an aero-thermodynamic viewpoint. Cycles for engines aiming at lower specific fuel consumption, improved power lapse rates, and implementation of contingency ratings are discussed. These cycles include nonregenerative and regenerative cycles. Design trends are presented for the major engine aerodynamic components.

1. INTRODUCTION

Gas turbine engines entered the helicopter propulsion arena in the 1950's because of their very attractive power-to-weight ratio. The helicopter's load-limiting characteristics were overcome by a substantial increase in specific power resulting from the low weight of the turbine engine. In addition, the technology of the turbine engine had become sufficiently advanced with respect to cycle temperature, cycle pressure ratio, and component efficiencies to offer an attractive fuel consumption. With the introduction of the light gas turbine engine, the industry had two basic options. The first was to maintain the same basic system performance with a lighter and, therefore, less expensive aircraft. The second option improved system performance through the increase in vehicle specific power. Industry chose the latter. The success of this decision can be seen in the rapidly expanding civil, as well as military, helicopter fleets. By 1973, approximately 70 percent of the over 10,000 helicopters in service were powered by the gas turbine engine. It has been estimated that by 1990 the western world will have over 23,000 military helicopters. During this same period, it has been projected that the civilian fleet will more than double to 26,000, thus passing the military.

This projected growth in the world helicopter marketplace has created a demand for new advanced, small engines. In this paper, we will review historical engine trends, discuss engine cycle considerations, and assess the current areas of engine component Research and Development (R&D) activity aimed at the new helicopter engines of the future.

2. HISTORICAL ENGINE TRENDS

The key features used in the first gas turbine engines (T53 and T55) specifically designed for helicopters in the 1950's are a combined axial-radial compressor and a free-power turbine with a front drive. These features adopted by most helicopter engine manufacturers are expected to be used in the next generation of engines ranging between 800 and 4000 horsepower. In the design of the early helicopter engines, an optimum balance was emphasized between compactness, performance, life, maintainability, and cost of ownership. Substantial improvements in all these areas were achieved by a steady evolution process illustrated in Figs. 1 and 2 which trace the history of the T53 and T55 helicopter engines over a period of 20 years in terms of specific fuel consumption and specific power. The output power of these engines was increased by a factor of two or more while specific fuel consumption was reduced by one third. Much of this improvement came as a result of increased cycle temperature, cycle pressure ratio, and better component efficiencies. Substantial increases in cycle temperature were accomplished by using a combination of improved materials and introducing cooling techniques applicable to small engines. The introduction of transonic axial compressor stages allowed the cycle pressure ratio and air flow rates to increase with minimum engine modifications.

During the 1950's, engine development was focused on weight and performance. The 1970's saw increased emphasis on the trade-offs between weight, performance, cost, and reliability. As a result, engines became simpler with improved reliability and efficiency. Fig. 3 illustrates these gains in simplicity and performance by comparing the original T53 engine with the newer LTS 101. Engine simplicity was achieved with significant performance improvements. The beginning of the evolutionary development phase of the LTS 101 is also shown. As can be seen on Figs. 1 and 2, this engine achieved these performance gains through higher cycle pressure ratios and increased turbine inlet temperature.

As we entered the 1980's, the industry became more sophisticated in the evaluation of engine progress. The concept of life-cycle cost gained widespread recognition, and all aspects of engine costs such as design, development, fuel usage, acquisition, and their impact on the helicopter life-cycle cost are now considered. Maintenance concepts such as modular construction have been designed into the engines. Performance projections for these newer, advanced-technology engines are also shown in Figs. 1 and 2. Continued progress under these ground rules, illustrated in Fig. 4, demonstrates the gains achieved by Lycoming's Advanced Technology Demonstrator Engine, which was sponsored by the U.S. Army. This engine, currently under development, was designed with life-cycle cost ground rules and, as such, recognizes the major impact of rapidly rising fuel prices.

In spite of the dramatic progress made in helicopter engine designs to date, it is anticipated that significant advances in performance will be achieved by engines of the future. This advancement will show up primarily in reduced fuel consumption and engine weight. The combination of these two effects will result in major increases in payload capability or range.

3. CYCLE CONSIDERATIONS

Mission utilization strongly affects the philosophy of engine design. Military and commercial applications that require longer missions and greater utilization need a more sophisticated engine having low fuel consumption and requiring less maintenance to achieve low life-cycle cost. Low acquisition cost is a stronger factor in the life cycle cost for aircraft having low utilization as in some general aviation applications.

3.1 Part-Power Fuel Consumption - Nonregenerative Cycle

Mechanical considerations determine the maximum allowable cycle temperature. For each level of cycle temperature, there exists a cycle pressure ratio that produces the minimum specific fuel consumption, as shown in Fig. 5, where specific fuel consumption (SFC) and specific power are shown to vary as a function of pressure ratio with temperature as a parameter. The optimum cycle pressure ratio that produces minimum fuel consumption increases with cycle temperature and component efficiency index.

There have been steady improvements in turbine cooling technology and high-temperature materials applicable to small engines. As a result, there has been a gradual increase in the allowable cycle temperature. Since compressors can achieve higher stage pressure ratios, there is an incentive to attain higher pressure ratio cycles with a reduction in the number of stages.

Two part-load lines shown in Fig. 5, indicate how the performance would vary if a constant component efficiency index (down to 60 percent part-power) could be achieved. By selecting a cycle with a pressure ratio of 20 and temperature of 2800°R compared with a cycle having a pressure ratio of 14 and a temperature of 2600°R, a seven-percent reduction in SFC at maximum power could be obtained. However, at 60-percent part power, the improvement in SFC is increased to 11 percent with the more advanced cycle. This improvement indicates that even though fuel consumption benefits are achieved at maximum power with advanced cycles even larger benefits result at part power where the greater portion of operation occurs.

3.2 Power Lapse Rate

A prime sizing condition for twin-engine helicopters is based on the "One Engine Inoperative" flight condition at 4000 feet altitude on a 555°R day. It then is beneficial to select cycle parameters that minimize the lapse in power caused by high ambient temperatures. The variation in power output of typical engine cycles was studied as the ambient temperature varied from 519° to 555°R. The relative influence of design parameters, such as cycle pressure ratio, turbine inlet temperature, and component efficiency index were included.

At high ambient temperatures, the specific power output from the power turbine is reduced as a result of the lower pressure ratio available. Fig. 6 shows specific power loss as a function of cycle pressure ratio and turbine inlet temperature for two component polytropic efficiency levels. For polytropic efficiencies of 0.8 and a turbine inlet temperature of 2250°R, the specific power loss resulting from hot-day operation increases from 11 to 23 percent as the cycle pressure ratio is increased from 8 to 20. At higher cycle temperatures, the specific power loss becomes less dependent upon cycle pressure ratio. At higher component efficiency indexes, this loss almost becomes constant.

In Fig. 7, the upper curve shows the reduction in airflow rate as a function of cycle pressure ratio when the ambient temperature increases from 519° to 555°R. The reduction in airflow is shown to be lower at low-pressure ratios than at high-pressure ratios. The loss in airflow increases 40 percent as the cycle pressure ratio varies from 5 to 20. The reduction of specific power (Δ shp/W_a) is again shown as a function of the cycle pressure ratio with the turbine inlet temperature as a parameter but for a polytropic efficiency index of 0.86. Generally, the loss in specific power decreases substantially with increased turbine inlet temperature. The total power loss is a combination of the airflow loss and the specific power loss. This combination increases with pressure ratio for a constant turbine inlet temperature. As previously shown, the simultaneous increase of cycle temperature with pressure ratio improves the specific fuel consumption. This simultaneous increase in cycle temperature with pressure ratio also minimizes the lapse in power due to hot-day operation as indicated by "Line A" in Fig. 7.

3.3 Contingency Ratings

The rapid gain in popularity of twin-engine helicopters has created a new demand on the engine designer to provide safe operation when one engine is inoperative (OEI). In this paper, we will restrict ourselves to the impact of this requirement on the selection and design of engine components. The problem is the airframe manufacturer's natural desire for a large amount of contingency power (as much as 50%) with few, if any, "use restrictions" and with no special maintenance actions required. Taken at face value, this means the engine must be significantly larger. Such an engine, however, would penalize the helicopter by being heavier, larger, more costly, and less fuel-efficient for a given mission.

The commonly accepted method of obtaining contingency power is to merely advance the throttle. This results in higher turbine inlet temperature and higher speed. Power increases of 10 to 20 percent can normally be achieved in this manner. But, consideration of even higher ratings eventually runs into fundamental limitations based on engine component sizing. These limitations are primarily concentrated on two components. The first component considered is the compressor. All compressors reach an operating condition called "choked-flow" where the engine airflow no longer increases in proportion to speed changes. Eventually, an absolute referred airflow limit is reached; this is caused by the choking of the front stages of the compressor. When this condition is reached, no further power increase can be obtained by airflow, and the compressor stages become mismatched, thus causing a drop in the surge line. At the same

time, the compressor efficiency decreases rapidly and causes the engine operating line on the compressor map to turn upward. Both effects then combine to continually decrease engine transient surge margin until steady-state surge is encountered.

An additional consideration involves the basic differences in the matching of turboprop and turboshaft helicopter engine components. In the turboprop, the engine sizing criterion is altitude climb and cruise where low ambient temperatures are encountered. The turboshaft helicopter engine, however, is normally much more concerned about hot-day lift capability and part-load efficiency under relatively low-altitude cruise conditions. Because of compressor referred-speed characteristics, an optimum helicopter turboshaft engine should be matched at a higher compressor referred speed at takeoff conditions than a turboprop engine; this reduces the maximum increase in airflow obtainable for a turboshaft engine before the compressor chokes. Consequently, an engine matched for maximum hot-day power capability will have a smaller increment of contingency power available than a turboprop engine.

The power turbine also plays a role in that a large power excursion for OEI use forces the turbine to operate at higher exit Mach numbers. This condition reduces the component's efficiency until it also reaches a choked condition. In addition, the power turbine must operate at a constant rpm because of the helicopter's rotor acceleration limitations. Operating at constant rpm results in increased exit swirl leaving the power turbine with a further reduction in potential power output. There are also mechanical problems associated with the higher gas producer rotor speeds and temperatures caused by advancing the throttle. These fundamental limitations can be overcome with proper design, but at the expense of having larger components.

From the standpoint of fuel consumption, it is also desirable not to just make a larger engine. As seen in Fig. 8, an engine that is oversized to cover the contingency requirements will suffer a five-to-ten percent penalty in specific fuel consumption under typical cruise power requirements. This is because the engine is effectively operating at a lower percentage of a design power. Based upon this effect, it is desirable to limit the amount and usage of contingency power so that this penalty is minimized.

Considering the above factors, trade-off studies should be conducted to establish the amount of contingency power required and to determine its impact on the helicopter system prior to final engine sizing. The desire to have 50-percent additional power available under an OEI condition does not appear to be practical when considering a "push the throttle" approach. Controlled usage of 10-to-20 percent additional power would be more realistic. Unfortunately, this does not entirely solve the problem since any contingency power margin designed into the engine will tend to be reduced as additional power is required during the inevitable growth of the system. A continuing program to develop future contingency ratings as the engine grows must be maintained after a system enters production.

3.4 Regenerative Cycles

In the past, regenerators have received recognition as to their theoretical potential for significantly reducing the fuel consumption of a gas turbine engine. But in recent years, they have progressed to a point where a more realistic appraisal can be made of their potential benefit in helicopter applications. Experimental regenerative engines have been tested in various vehicles and have even been flown experimentally in a helicopter. The U.S. Army's Abram tank now in production is using a regenerative engine. Studies were recently sponsored by the U.S. Army at Fort Eustis, Va., to investigate the potential of using such engines on helicopters in the future. A summary of regenerator design and performance characteristics will be discussed in the Component Section of this paper. This discussion of regenerators will focus on identifying various approaches to their possible use on modern helicopters.

The basic problems involved with the use of regenerators on helicopters are weight and size. Although reductions in specific fuel consumption of twenty percent or more are readily achievable with today's heat exchanger technology, the weight impact of such units severely penalizes the performance of the weight-sensitive helicopter. Performance benefits of the aircraft system depend on a trade-off between the weight of the fuel saved and the increase in aircraft weight caused by the weight of the regenerator. In general, longer missions make the use of a regenerator more attractive. Several approaches that have the potential to reduce this weight penalty are discussed below.

The performance (effectiveness) of a regenerator is largely a function of its heat-transfer surface area, which in turn establishes its weight. Unfortunately, the physics of the heat-transfer process makes the relationship between effectiveness and weight nonlinear (see Fig. 9). One potential method to reduce the weight penalty is to size the unit for cruise conditions; this gives the benefit of part-load fuel savings at significantly reduced weight. A necessary addition, however, would be the use of a gas-side bypass valve at high power to reduce the pressure loss and back pressure on the power turbine at high flow rates. Other approaches include the use of one regenerative engine on a twin-engine helicopter. Cruise power could be taken from the regenerated engine while the other engine is either shut down or operated at a low idle setting.

Additional fuel savings can be realized in a regenerated engine by using a variable-area power turbine. This device shifts the point of minimum fuel consumption to the desired part-power operating condition. A typical operating line, shown dotted on a regenerative cycle plot (Fig. 10) indicates how the fuel consumption varies at part-power operation. The specific fuel consumption and specific power are shown versus cycle pressure ratio with cycle temperature as a parameter for a constant component efficiency index. "A" denotes the cycle pressure ratio and temperature at maximum power. Closing the power turbine stator while maintaining a constant turbine inlet temperature causes the pressure ratio across the gas producer turbine to decrease. This action results in a reduction in the work output from the gas producer turbine, thereby requiring the compressor to decrease in speed, flow, and pressure ratio. The lower flow results in decreased power even though the turbine inlet temperature is maintained (line A-B). Point "B" denotes the optimum cycle at which a further reduction in power is obtained by lowering the cycle temperature (line B-C). In this manner, a nearly constant specific fuel consumption is achieved down to part-power where a greater portion of cruise operation occurs.

4. AERODYNAMIC COMPONENTS

Component research and development activities are continuing to enable the advanced cycles to be achieved. This section discusses current approaches to obtain improved components for future helicopter powerplants. The discussion is limited to the prime aerodynamic components including compressors, combustors, turbines, and regenerators.

4.1 Compressors

Substantial advances have been made in compressor technology over the past ten-to-fifteen years. While peak efficiencies have not increased significantly, these efficiency levels have been extended to increasingly higher cycle pressure ratios. Much progress has been accomplished in the analytical techniques applied to compressor design and development. Highly transonic and fully supersonic stages have been developed to the level of performance necessary for use in advanced engines. Figure 11 shows the measured performance map of a small single-stage axial compressor, designed for a five-pound per second airflow, operating at a stage-pressure ratio of 2:1. The single compressor stage shows excellent characteristics with the peak efficiency contours located well away from the surge line. This stage has been successfully coupled with a fully supersonic second stage to produce an overall pressure ratio of approximately 3.5:1. Earlier technology would have required five or six stages to accomplish the same pressure rise.

Major development efforts have been directed toward the centrifugal compressor. Improved theoretical modeling of this unit has increased its range of useful pressure ratios. Analytical techniques from transonic axial states have been applied to the design of the impeller's inducer section. Improved structural analysis has allowed higher tip speeds. In addition, the use of leanback on the impeller vanes has improved diffuser performance by reducing the entering Mach number. Figure 12 shows a typical selection of modern centrifugal impellers. Pressure ratios have risen from the 3:1 range to the level of 10:1 with experimental programs directed toward 15:1. As a result, the compressor section of a typical high-pressure ratio advanced helicopter engine may consist of one or two axial stages, plus a high-pressure ratio centrifugal stage, as opposed to the 10-to-14 axial stages used in older engines.

Improved operating range (surge margin) has been achieved through the use of long chord blades in the front stages of axial compressors. Rotor tip treatment has also proven useful in increasing surge margin in a simple, reliable manner.

As a result, it can be expected that improvements in engine performance will continue due to the higher cycle pressure ratios available to small engines. Even though the complexity of the compressor has been reduced dramatically, excellent levels of efficiency have been maintained.

4.2 Combustors

The modern gas turbine combustor, faced with a rapidly changing environment, has had to adjust to higher cycle pressure ratios and temperature levels. Higher air temperature leaving the compressor has made cooling of the liner more difficult. In addition, the combustor designer faces the problem of exhaust emissions that are coupled with the demand to operate with a wider range of new types of fuel. The necessary solutions are particularly demanding with the small combustors typical of helicopter powerplants.

New combustor design and development procedures are enabling these challenges to be met successfully. Great effort is being devoted to the development of accurate analytical models of combustion systems. Overall design approaches, using one-, two-, and even three-dimensional representations, are being developed and used. Interactive design methods have been started. More detailed representations developed for specific purposes are also making steady progress. Fuel-spray modeling has yielded performance benefits and has been successfully coupled with experimental techniques to verify its accuracy. Liner-wall analysis techniques are particularly useful in the development of long-life combustors because of the high-temperature cooling air and the higher combustor exit temperatures of advanced engines. Characteristic time-modeling procedures have yielded benefits in reducing exhaust emissions. Fundamental modeling of the chemical processes is being accomplished by using zone combustion models that analyze various steps involving droplet evaporation, mixing, and the chemical reaction. The successful application of these methods of analysis are yielding good solutions using simpler hardware.

An example of the successful application of these analyses is in the area of exhaust emissions. The basic problem involves achieving a satisfactory trade-off between the high-power region where smoke and the oxides of nitrogen (NO_x) predominate and low power where carbon monoxide (CO) and unburned hydrocarbons (UNH) manifest themselves. From a combustor design viewpoint, this requirement presents conflicting demands for high- and low-power operation.

The solution in large engines was to develop staged combustion systems where literally a small combustor is used in the low-power regime and an adjacent, larger one used at high-power levels. The size, cost, and complexity of such approaches were undesirable for small engines. Figure 13 depicts an annular combustor designed for small helicopter engines. This circumferentially stirred combustor admits primary combustion air through slots in the liner header to produce a flow vortex about a circumferential mean line. Folding air jets enter through the inner wall to reinforce this primary zone recirculation pattern which is then forced to turn axially on either side of these folding jets to create a pattern in the shape of a horseshoe. A single fuel injector supplies both legs of the horseshoe pattern, thus reducing by a factor of two the number of fuel injectors required. This concept introduced a major cost benefit and also resulted in doubling the size of the small fuel nozzle passages. Testing of this concept verified a reduced NO_x production at high power levels where NO_x is most critical. As a result, combustor modifications could then be made to reduce CO and unburned hydrocarbons (UNH) at idle. The tested results for a baseline combustor and a low emissions version of the same design are shown in Fig. 14 and compared with the now abandoned United States EPA standards of 1973 for an equivalent turboprop version of a small helicopter engine.

Continued development of advanced combustor modeling technique will lead to similar improvements in combustor life, exit temperature distribution, and the ability to handle a wide range of fuels.

4.3 Turbines

The design of a modern high-temperature turbine requires the integration of several technologies that include aerodynamics, heat transfer, and materials.

All of these technologies play an important roll in producing a design that is efficient and possesses good mechanical integrity. The trend in turbine design is towards higher temperature and higher work extraction per stage; this not only reduces the number of stages but also reduces the gas temperature entering the next stage. As a result, cooling air requirements are also reduced. This higher work extraction, however, requires increased rotational speeds that result in higher stress levels.

Size effects are critical to highly efficient, small turbines. Figure 15 illustrates the small size of the gas producer turbine for an advanced 800-horsepower engine. Dimensional effects, such as tolerances, clearances, surface finish, leading and trailing edge thicknesses, and casting wall thickness, cannot be scaled down. Therefore, these effects influence small turbine losses to a much greater extent than large turbines. Since blade chords are not scalable, aspect ratios become smaller, thereby resulting in larger secondary losses. Cooling airflow requirements are larger because of disproportionately higher wetted turbine surface area. This additional cooling flow not only creates more losses in the turbine but is detrimental to the cycle. The challenge then is to minimize these effects with proper design. Some turbine design features, which are presently being pursued, are discussed below:

The gas generator turbine for the LTS 101 turboshaft engine is an example of a good aerodynamic design for its size class (5 lbm/sec) as reflected by the tested performance map shown in Fig. 16. Careful control of the surface velocities on the blades and vanes has contributed substantially to the high efficiency. In addition, tip leakage losses have been minimized by designing for a cylindrical tip diameter and moderate tip reactions.

Blade tip clearances strongly affect the end-wall losses and are magnified in small turbines. Special attention was given to cooling the cylinder surrounding the high-temperature gas producer turbine so as to minimize the "out of round" distortion to insure control of the tip clearance and leakage and to extend its life. Figure 17 shows the design of a typical cooled gas-producer turbine cylinder with clearance control. Cooling air is channeled to the front of the cylinder where it enters a labyrinth of five passages. The passages, which are interconnected, allow air to flow circumferentially as it passes from one passage to another towards the back of the cylinder where it reenters the main gas stream. The purpose of the clearance control is to safely maintain tight clearances during various operating conditions, i.e., takeoff, cruise, acceleration, and deceleration. This clearance control results in a reduction of the end-wall losses with a gain of several percent in turbine efficiency.

The cooling air network for a gas producer turbine is shown in Fig. 18. Note that the heat shield has a dual purpose: first, it insulates the turbine cooling air as it passes along the combustor liner; and secondly, it allows some cooling air to pass between the shield and liner to provide convective and film cooling to the liner.

The cooling air preswirl nozzles (shown in Fig. 18) are used to extract energy from the high-pressure cooling air, thus reducing its pressure and temperature before it enters the gas producer rotor. The lower temperature benefits cooling of the turbine disk and blades.

An approach to the design of a high-temperature rotor blade is depicted in Fig. 19. Cooling air enters the center of the blade and is allowed to impinge on the external skin through side holes. This approach results in high heat transfer coefficients and effective cooling. The air then passes through film cooling holes to provide a protective film surrounding the blade.

New blade materials and casting processes are constantly being explored. Blades are being manufactured by directionally solidifying blade material to produce a columnar grain structure that improves its stress-rupture strength characteristics. Single crystal blades are also being produced for increased strength by eliminating the grain boundaries. Directionally solidified blades permit a 35°F higher temperature, whereas single crystal blades can tolerate a 130°F increase in temperature for the same stress level and blade life.

As discussed above, regenerative engines benefit substantially from variable turbine geometry so that high-cycle temperatures are maintained at part power, thereby resulting in lower specific fuel consumption. Although variable geometry has been limited in the past to uncooled power turbines, it should be extended to cooled turbines so that adequate compressor surge margin can be insured at part power in order to achieve maximum efficiency. It is by these means that the turbine can be continuously altered to meet power or acceleration requirements.

As discussed above, the turbine component requires considerable attention to a variety of details that will contribute to improved performance and mechanical integrity. These details require the integration of the various disciplines to produce a successful design.

4.4 Regenerators

Recent studies have shown that regenerative engines designed for helicopter applications using present state-of-the-art components can achieve a 20-percent fuel savings over nonregenerative type engines. The combined engine and fuel weight of a typical regenerative engine is approximately the same as a nonregenerative engine configured for missions of two to three hours. Fuel savings offset the higher acquisition cost of a regenerative engine for fuel cost above $1.80/gallon (1979 dollars). It is envisioned that by 1985 a small regenerative helicopter engine can become economically feasible. This is especially true since it is becoming more difficult to make advancements in component efficiency, and we may have to resort to other means, such as regeneration, to achieve future reductions in fuel consumption.

The regenerative engine now in production for the M1 Abram tank has provided valuable experience in the design, development, and manufacture of regenerators that could be applicable to helicopter turboshaft engines. Figure 20 is a schematic of this "multiwave-plate" regenerator core. Hot gas enters the core along the inner annulus, passes radially through the heat transfer surfaces, and then exhausts. High-pressure air enters the core through the triangular passages and then enters the heat-transfer surfaces in a cross-flow direction, with respect to the exhaust gas. This air then flows radially inboard (counterflow to the gas) and flows once again in a cross-flow direction and returns to the combustor through the elliptical passages.

Preliminary designs of tube-type regenerators have also been made. An example of such a design is illustrated in Fig. 21, where a regenerative LTS 101 is depicted with a tube-type regenerator. The regenerative engine is identical to the nonregenerative version except for the combustor housing. The housing assembly incorporates the following functions by which it:

1. Provides a diffuser and duct for air leaving the compressor passing to the regenerator

2. Provides a duct for the hot air leaving the regenerator passing to the combustor

3. Provides a main structural member to attach the regenerator to the engine midframe at the compressor diffuser exit; this is the main rear support for the rear gas producer rotor bearing, the power turbine assembly, and the fuel manifold.

The regenerator which is a cross-flow, tube-type heat exchanger having two passes on the tube (air) side and one on the shell (gas) side was designed to achieve 72-percent effectiveness with an air-side core pressure drop of two percent and a gas-side core pressure drop of five percent. The thermodynamic details are given in Fig. 22. The tubes are brazed to the forward header, rear header, and two intermediate baffles. The baffles are extensions of the hot-gas exit diffuser that improves the flow distribution over the tubes, as well as providing a tube support structure.

A comparison of the part-load performance of the regenerative engine with a nonregenerative engine is presented in Fig. 23. The regenerator pressure drop, along with the additional diffuser loss and leakage, caused a reduction in maximum power. The specific fuel consumption is reduced 20 percent at 50 percent power which is approximately a cruise power setting for helicopter applications.

The regenerative engine has progressed to a point of becoming economically attractive for helicopter applications. Further development of manufacturing methods such as welding processes, plate forming, tube extrusion, and fabrication is required to reduce regenerator manufacturing costs.

5. CONCLUSIONS

Helicopter engines of the next decade will exhibit significantly improved performance if current Research and Development programs are pursued to their full potential.

Small nonregenerative turboshaft engines in the 800-to-1200 horsepower class are projected to have cycle pressure ratios in the range of 18 to 22:1 and turbine inlet temperatures of about 2800°R. Specific fuel consumption will approach 0.4 lbm/hp/hr.

Compressors used in future engines will achieve this cycle pressure ratio by using fewer stages. Configurations consisting of a combination of axial and centrifugal stages or dual centrifugal stages are viable approaches. The centrifugal compressors will have leanback impellers to improve efficiency and range. The turbine will be aerodynamically, highly loaded with active clearance control to minimize tip clearance and secondary losses. Advanced materials such as single crystal blades will be used to extend life and reduce cooling requirements. Environmental controls will become more severe requiring that more emphasis be placed on developing combustors having low emissions. And finally, because of the escalation of fuel prices, the regenerative turboshaft engine will receive more attention for helicopter applications with long duration mission requirements.

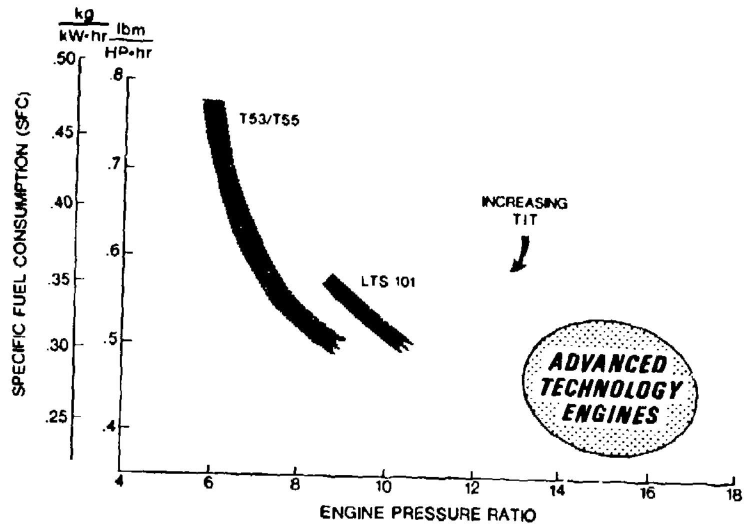

Figure 1 Historical Improvement in Fuel Consumption

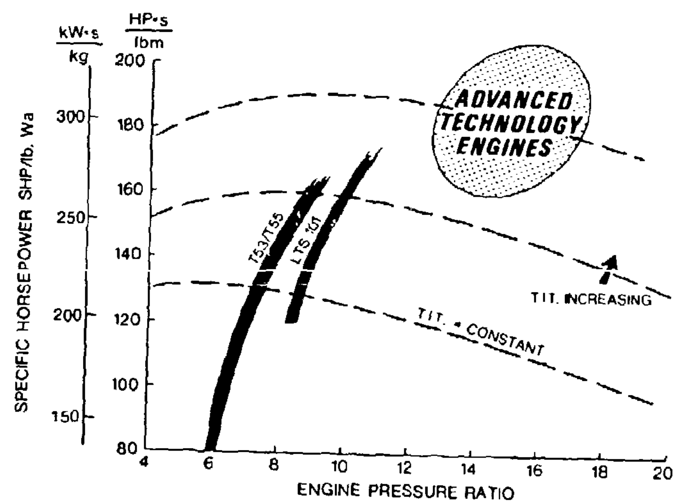

Figure 2 Historical Improvement in Specific Power

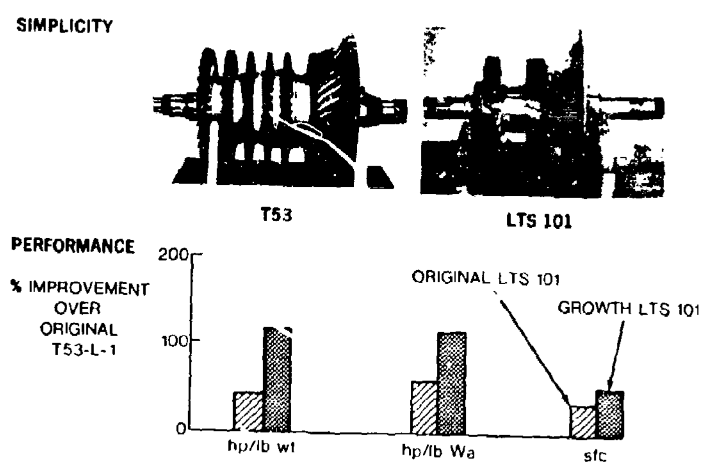

Figure 3 Simplicity Achieved with New Centerline

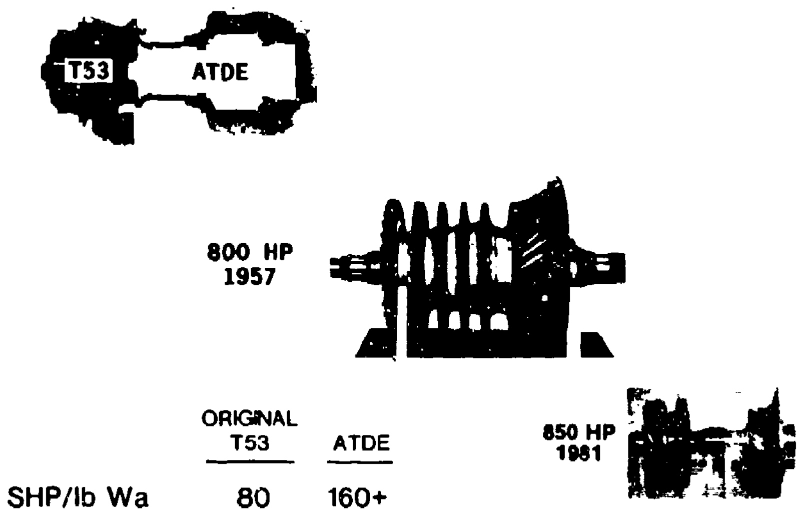

Figure 4 Advanced Technology Demonstrator Engine

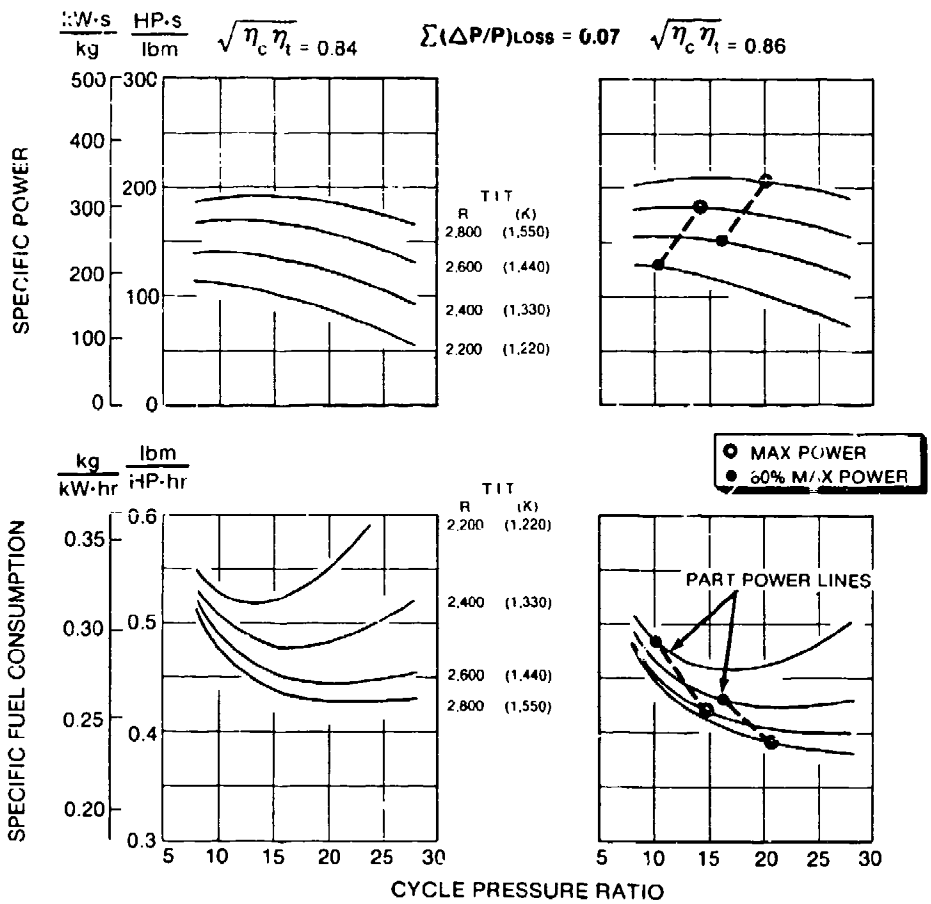

Figure 5 Turboshaft Engine Performance - Nonregenerative Cycle

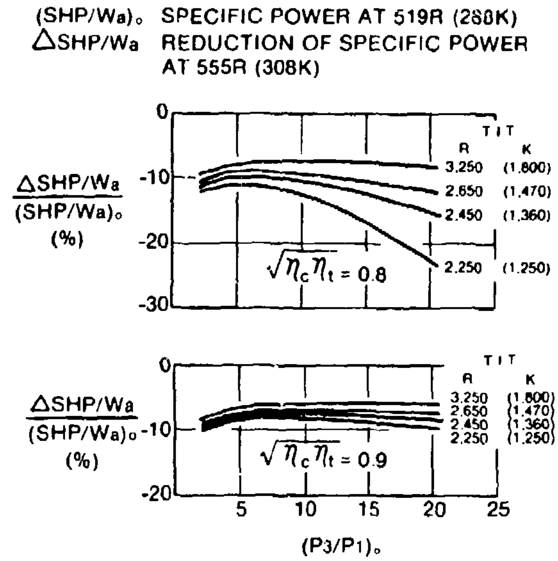

Figure 6 Effect of Cycle Parameters on Hot Day Specific Power Reduction

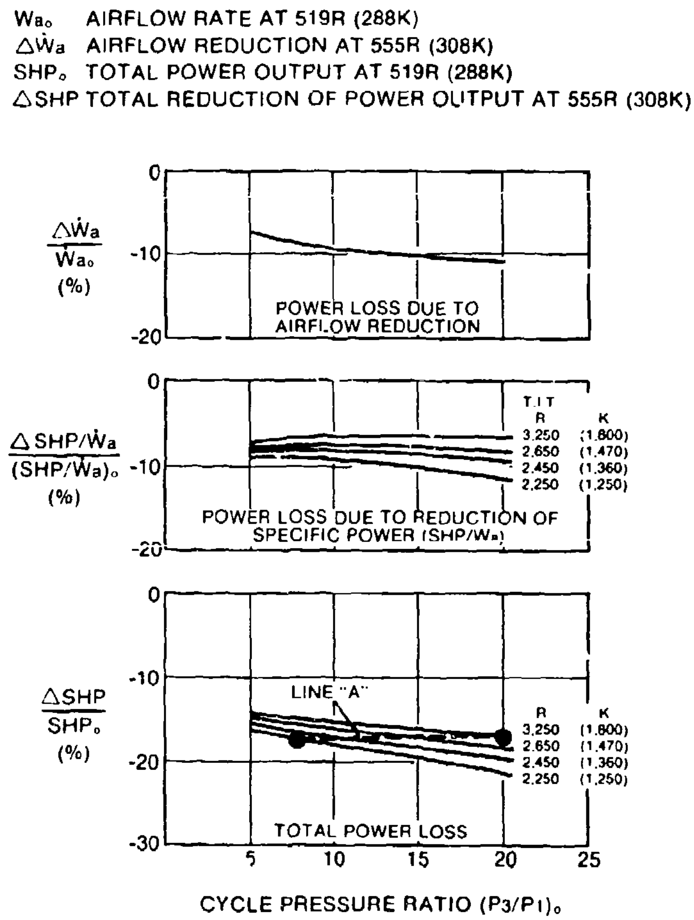

Figure 7 Effect of Cycle Parameters on Hot Day Engine Power Loss for $\sqrt{\eta_c \eta_t} = 0.86$

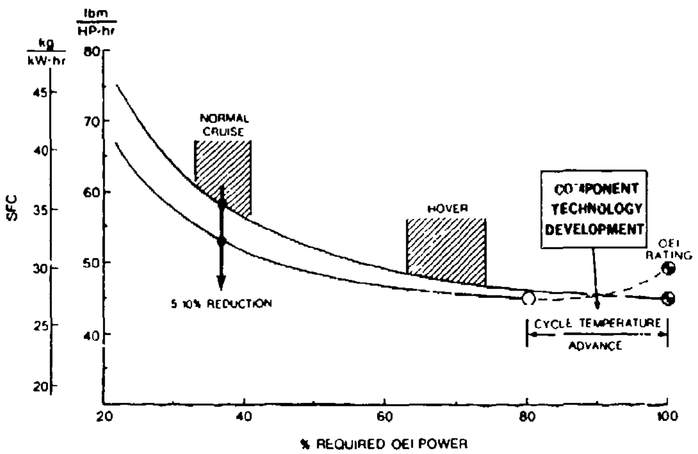

Figure 8 Impact of "One Engine Inoperative" Sizing on Twin Engine Installation

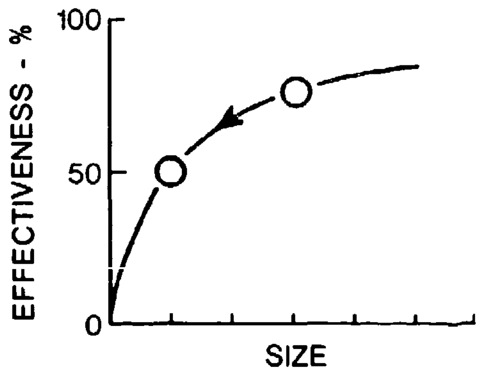

Figure 9 Aircraft Regenerator Effectiveness vs Size

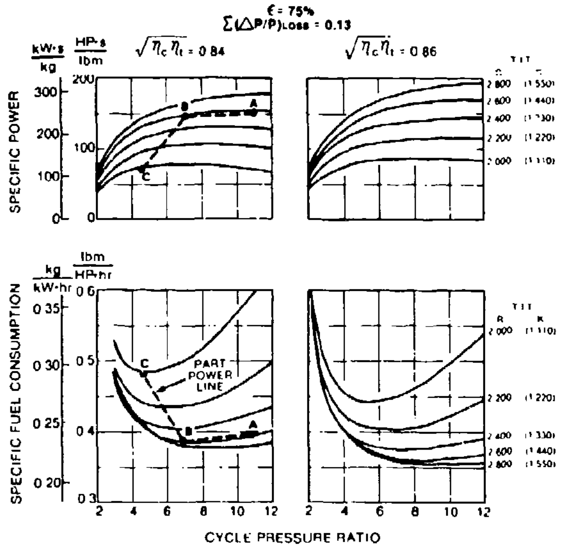

Figure 10 Turboshaft Engine Performance - Regenerative Cycle

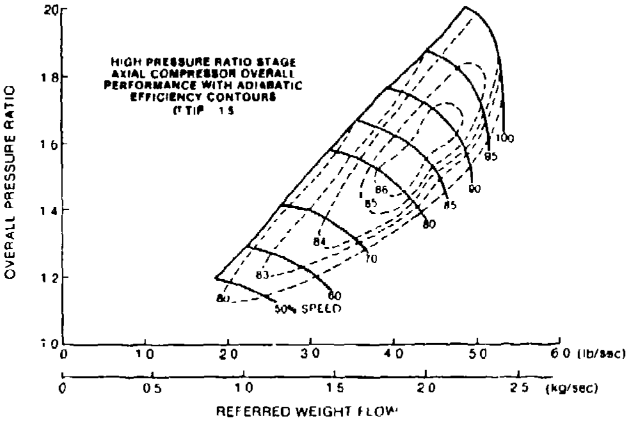

Figure 11 High Pressure Ratio - Single Axial Compressor Measured Performance

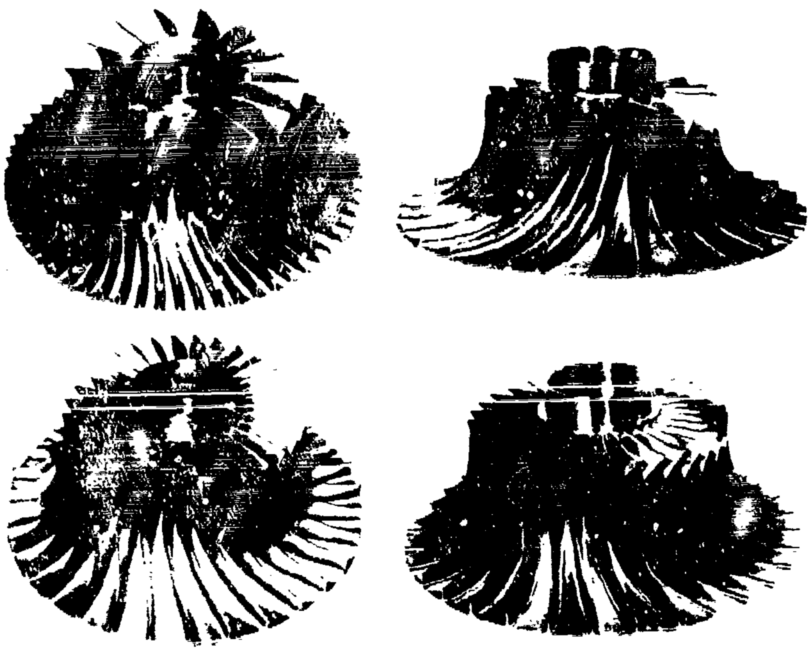

Figure 12 Typical Research Impellers

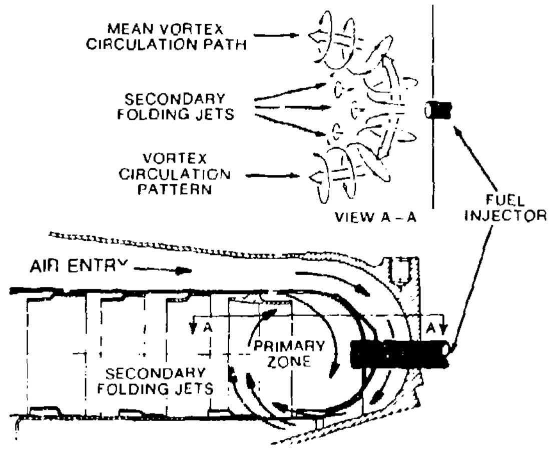

Figure 13 Folded Annular Combustor - "Horseshoe" Flow Pattern

	UHC	CO	NO_x	SMOKE NUMBER
Goal*	4.9 (3.0)	26.8 (16.3)	1[illegible] (7.[illegible])	50
Baseline	10.8 (6.6)	36.7 (22.3)	4.2 (2.6)	47
Low Emissions Combustor	2.5 (1.5)	15.8 (9.6)	5.5 (3.3)	16

* lbm/1,000 hp/hr/cycle (g/kW/hr/cycle)

Figure 14 Emissions Test Results for "Horseshoe" Combustor

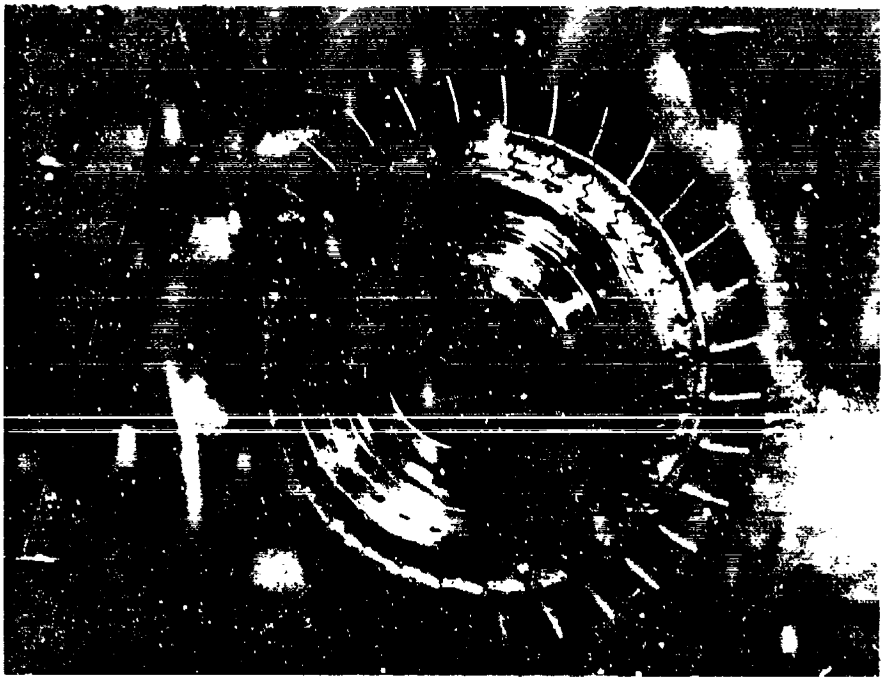

Figure 15 Advanced Gas Generator Turbine

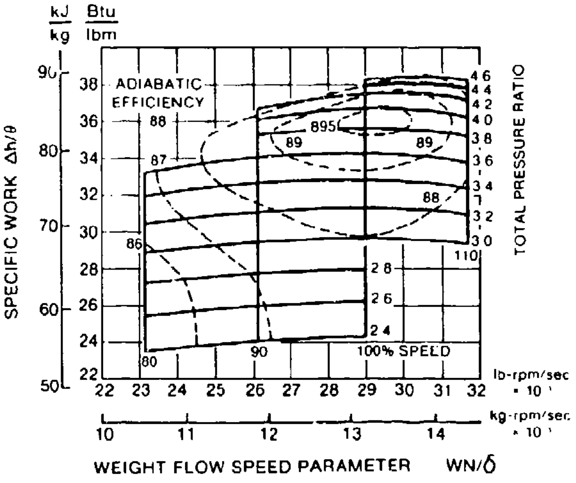

Figure 16 Tested LTS 101 Gas Generator Turbine Performance Map

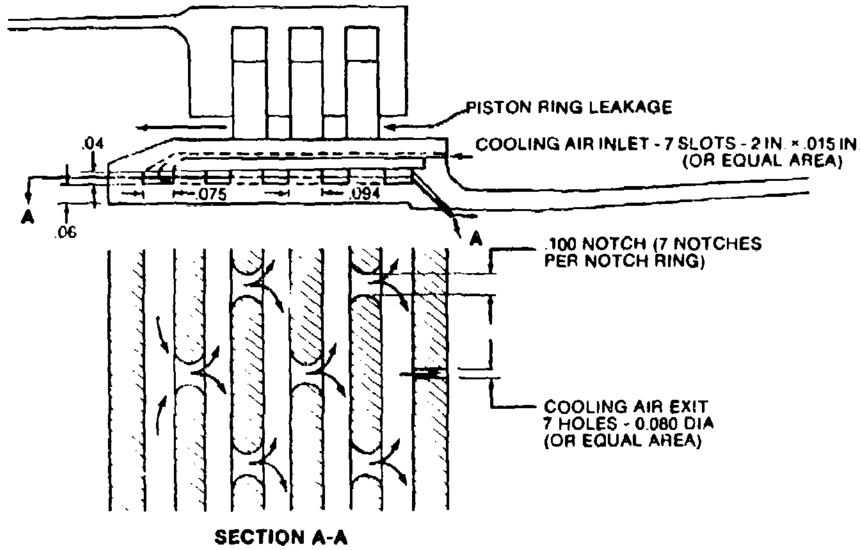

Figure 17 Gas Generator Turbine Cooled Cylinder

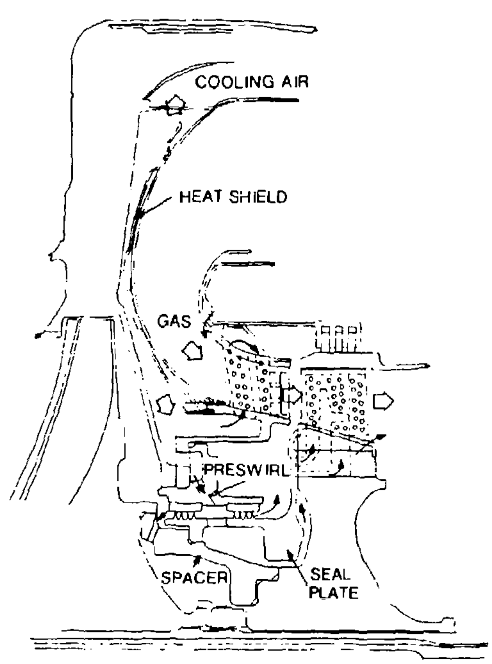

Figure 18 Gas Generator Turbine Cooling Network

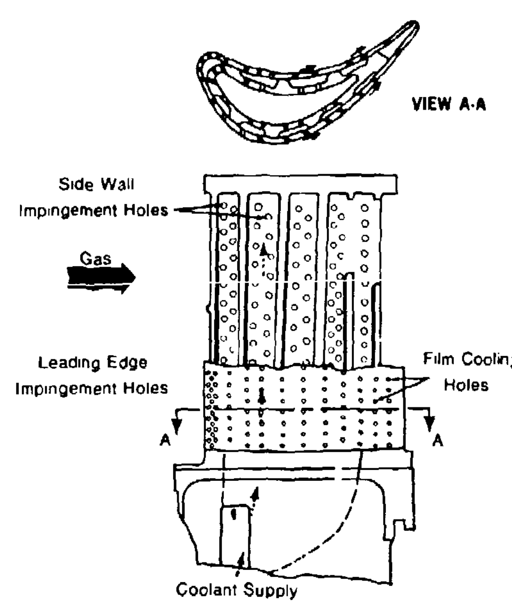

Figure 19 Small High Temperature Turbine Blade

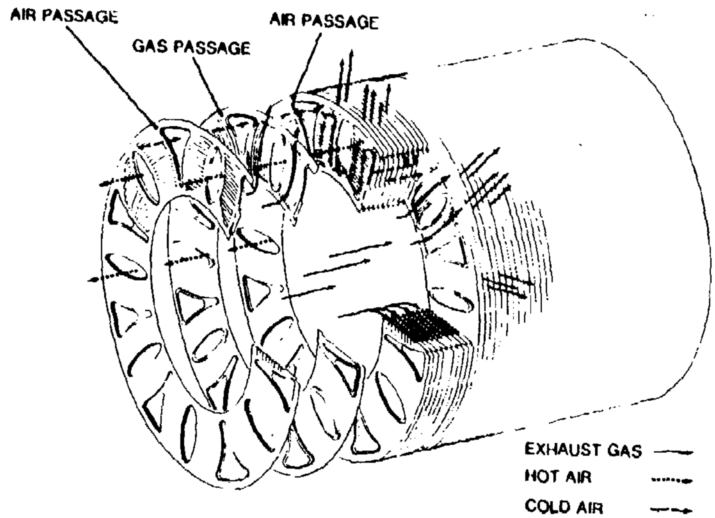

Figure 20 Multi-Wave Plate Regenerator Core

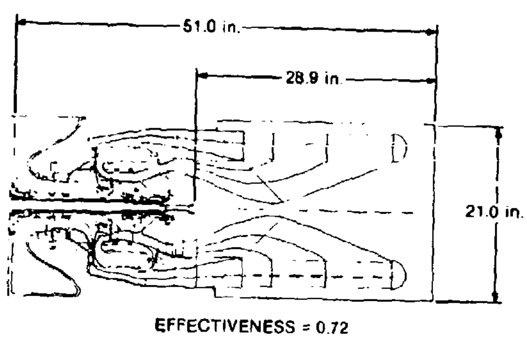

Figure 21 Regenerative Aircraft Turboshaft Engine in the 850 Horsepower Class

PERFORMANCE	
EFFECTIVENESS	0.72
NTu	3.3
$(\Delta P/P)_{air}$	0.02
$(\Delta P/P)_{gas}$	0.05
CORE DIMENSIONS	
NO. OF TUBE PASSES	2
NO. OF TUBES	9,020
TUBE DIAMETER	0.1 in.
TUBE WALL THICKNESSES	0.006 in.
TUBE (AIRFLOW) LENGTH	18.0 in.

Figure 22 Design Parameters for an Aircraft Engine Regenerator

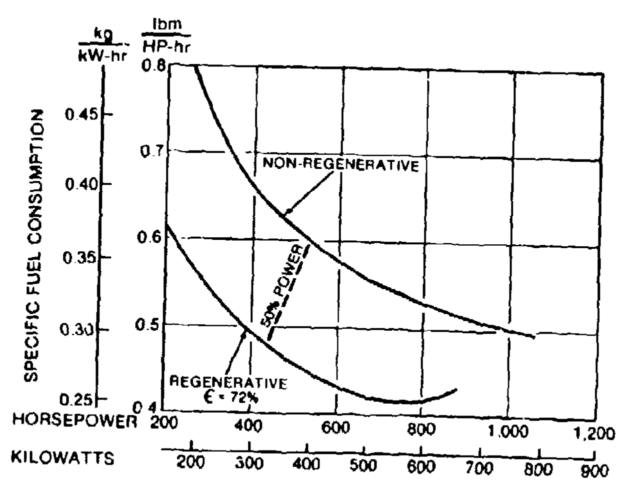

Figure 23 Engine Performance Comparison. Regenerative vs Nonregenerative

DISCUSSION

W.Heilmann, Ge

In relation to your high pressure turbine performance map, is it from a cold rig test?

Author's Reply

Yes.

W.Heilmann, US

Then, what would you expect the reduction in efficiency to be due to cooling?

Author's Reply

When operating in a hot environment with cooling, the efficiency would decrease between one and two points depending on the clearance control.

H.Kreiner, Ge

Do you have effective fabrication techniques to match the advanced tip clearance control and cooling designs?

Author's Reply

Fabrication techniques would be developed concurrently with the advanced blade design. Fabrication techniques already exist for the tip clearance control.

REGENERATIVE HELICOPTER ENGINES –

ADVANCES IN PERFORMANCE AND EXPECTED DEVELOPMENT PROBLEMS*

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Summary:

On the basis of modern engine component technology, including and emphasizing recent progress in high-temperature heat exchanger technology, a conventional and a regenerative helicopter engine in the 900 kW power class are compared in view of design, performance and life-cycle costs.

The comparison shows that the installation properties of the two engines are roughly the same. For a typical attack helicopter mission, the composite of power system weight (including the IR suppressor) and fuel/mission weight will be noticeably lower in the case of the regenerative engine. This is mainly a consequence of the latter's better fuel consumption.

The variable power turbine, being an indispensable part of a regenerative engine, at the same time leads to an extremely favourable transient behaviour with a moderate influence of severe cyclic loading on hot-part lifetime. No significant difference exists in IR emission at part load, due to the design of the IR suppressor for the same exhaust temperature at max. power.

The life-cycle costs of a future fleet of attack helicopters equipped with regenerative engines can be lower than with conventional engines. This holds true in spite of higher development and production costs of regenerative engines, if the present trend of growing fuel costs continues.

Finally, the problems which will have to be overcome are adequately described.

Symbols

H	m	altitude
M	kg/s; kg/h	mass flow
P	kW	power
p	bar	pressure
R	-	pressure ratio
SFC	$\frac{g}{kW \cdot h}$	specific fuel consumption
T	K	temperature
V	m^3	heat exchanger matrix volume
W	kg	weight
ε	-	heat-exchanger effectiveness
ϕ	$\frac{kg \cdot \sqrt{K}}{s \cdot bar}$	reduced mass flow $\frac{M \cdot \sqrt{T_t}}{p_t}$

Abbreviations or indices

C	compressor
F	fuel
HE	heat exchanger
IRS	infrared suppressor
LCC	life-cycle costs
t	total

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1. Introduction

Advanced attack and transport helicopters will require greater installed power than conventional helicopters with the same take-off weight, because of more stringent requirements as to their maximum flight speed and manoeuvrability. With twin-engined helicopters, moreover, in the event of failure of one engine at low altitude, the helicopter must have a high probability of being landed safely by the provision of a high emergency power with quick response characteristics of the other engine. The high power, which has to be installed for these reasons, means that for the major part of the mission the engines will operate in the low part-load range. Fig. (1) shows the requirements of an attack and a transport helicopter.

With conventional helicopter engines, the low part-load range signifies operation with unfavourable specific fuel consumption for a large part of the mission. In contrast, this is highly convenient for an engine equipped with a heat exchanger, which attains its minimum specific fuel consumption at roughly medium part load. Taking an attack helicopter as an example, an investigation is made into what design data are optimal for an engine fitted with heat exchanger and into how the heat exchanger must be designed and laid out, in order to achieve an improvement over a conventional engine with regard to lowest possible overall weight (weight of engine plus weight of mission fuel). In addition, the life-cycle costs of the engine with heat exchanger are contrasted with those of a conventional engine, and the two engines are compared concerning their necessary installation cross-section and installation length.

2. Optimization of the engine design data

Fig. (2) shows the specific fuel consumption at the design point and at 46% load (corresponding to the mean power; see Fig.1) in relation to the design pressure ratio. For the engine without heat exchanger (reference engine), the turbine entry temperature at the design point is 1400 K. A somewhat higher turbine entry temperature of 1430 K was assumed for the engine with heat exchanger (regenerative engine) but this has still to be explained. Accordingly, a maximum compressor pressure ratio of 12.5 is the optimum for the engine without heat exchanger, both at part load and full load. The same pressure ratio for the regenerative engine with 50 - 70% heat exchanger effectiveness also leads to the minimum specific fuel consumption at part load. Hence it is reasonable for an engine of this power class (P = 900 kW) to select the same compressor pressure ratio as in the reference engine.

The course of different temperatures and of the power turbine capacity at part load, important for the control of the regenerative engine, is shown in Fig. (3). Also given are the corresponding values of the reference engine, which are influenced by the pilot's lever only. Whereas the turbine entry temperature in the reference engine falls at part load the turbine entry temperature of the regenerative engine is held constant right down to 37% load with the aid of a two-stage variable power turbine. The turbine entry temperature is reduced below 37% load bearing in mind the uncooled power turbine. In addition to a favourable specific fuel consumption at part load, this control ensures that the mean temperature of the compressor-turbine blades (the characteristic of the mean blade temperature of the first stage is represented) decreases by about 50 K between full load and 20% power. Similarly, the mean temperature of the heat-exchanger matrix in this power range remains virtually unchanged, which means that the heat content of the heat-exchanger matrix changes only slightly. This results in the heat-exchanger matrix not giving rise to any thermal hysteresis during a deceleration or acceleration phase.

The turbine entry temperature was set with regard to the particular requirements of the attack helicopter mission, with consideration being given to the widely different thermal operating behaviour of the engines with or without heat exchanger. Fig. (4) shows the design parameters relevant to the permissible turbine entry temperature. The virtual avoidance of operational "thermal cycles" as a result of the "isothermal operation" in the case of the regenerative engine allows an increase in maximum turbine entry temperature in comparison with the reference engine. On the other hand, the longer operating time at approximately constant, i.e. higher, turbine entry temperature in case of the regenerative engine results in a reduction in the acceptable creep life. As a consequence, an additional 30 K in favour of the regenerative engine may be claimed for the same lifetime.

Fig. (5) shows the specific fuel consumption at the design point and part load for the reference engine and the regenerative engine. The compressor pressure ratio of both engines at the design point is 12.5; the turbine entry temperature of the reference engine 1400 K and that of the regenerative engine 1430 K. The heat exchanger effectiveness was varied between 50 and 70%. The significant improvement in the specific fuel consumption of the regenerative engine relative to the reference engine in the predominant mission power range is apparent. With the above-mentioned heat exchanger effectiveness the improvement lies between 20 and 30% at medium power.

3. Heat exchanger selection and optimization

In the selection of the heat exchanger particular attention was paid to the exchanger's frontal area with regard to the arrangement of the heat exchanger behind the engine and the requirement not to increase the engine's maximum cross section.

Fig. 6 shows a sketch of the recuperators under consideration (tube, profile and plate-fin heat exchanger) to explain the arrangement and the flow pattern. The profile heat exchanger, undergoing development at MTU, is derived from the tube heat exchanger. Both recuperators have a 2-pass cross counterflow arrangement in contrast to the counterflow type plate-fin recuperator. The required matrix volume, matrix weight, frontal area and length are compared in Fig. 7 for a heat exchanger effectiveness of 60% and a total pressure loss within the matrix (air and gas side) of 7%. Worthy of mention is the low matrix volume of the profile heat exchanger. This applies to both the net and gross volume (the gross volume includes the space required for ducting air to and away from the heat exchanger). The matrix weight of the three recuperators under consideration is practically the same, whilst the profile heat exchanger has the smallest frontal area. (For defining the frontal area it should be mentioned that the flow area in front of the heat exchanger corresponds to the free flow area available inside the heat exchanger matrix (gas side).) With regard to installation length, the profile heat exchanger compares very well with the plate-fin heat exchanger. A great advantage with the profile heat exchanger, on the other hand, is the much simpler conductance of air to and from the heat-exchanger matrix in comparison with the plate-type model. Finally, because of their design (free thermal expansion of the U-tubes or U-profiles), both the profile and tube heat exchanger are particularly suited for high operating temperatures and thermal shocks, whereas the plate-fin heat exchanger may be considered sensitive with regard to local temperature differences and thermal shocks. Therefore for purposes of the following optimization the profile heat exchanger was selected, as all-in-all this model best meets the requirements for favourable installation in a helicopter engine.

The optimization of the profile heat exchanger was carried out with regard to minimal overall weight of the power system (engine plus heat exchanger plus fuel per mission). The weight of the required infrared suppressor and ducting of air and gas to and from the heat exchanger was not included as variables in the optimization process. From earlier investigations it was known that these weight contributions have only a slight influence on the optimization of the heat exchanger dimensions. Fig. 8 shows the matrix weight, engine weight and weight of fuel required for a mission in relation to the heat exchanger effectiveness. The weight of the engine was matched for a constant design output of 900 kW, with the reference engine being used as the starting point. Computation of the amount of fuel required for a mission was based on a typical mission characteristic of an attack helicopter. The range in weight for any given heat exchanger effectiveness derives from various matrix pressure losses. The minimum overall weight is obtained in the range of 55 - 65% heat exchanger effectiveness.

4. Gas temperature at the engine exhaust

The turbine entry temperature of the reference engine is compared with that of the engine with heat exchanger at part load in the left-hand half of Fig. 9, based on Fig. 3. The turbine entry temperature is held constant over a wide part-load range with the aid of a two-stage variable power turbine. In the part-load range this control leads to virtually constant exhaust temperature downstream of the infrared suppressor. In contrast to the reference engine, for mean power (46% power) this gives rise to an increase in exhaust temperature downstream of the IR suppressor of 35 K. Accordingly, with the precondition chosen here (same full-load exhaust temperature at the outlet from the IR suppressor) no appreciable difference in the radiated infrared energy is to be expected. However, this takes for granted that the heat exchanger assembly is very efficiently insulated with simultaneous saving in weight as opposed to heat radiation.

5. Transient behaviour

A further advantage of the design with variable power turbine lies in the reduction in the acceleration time of the gas generator. A gas-generator acceleration time of 1.0 - 1.5 seconds can be attained by opening the power turbine fully during acceleration. This contrasts with the 3.0 - 3.5 seconds for the reference engine. At the same time, the maximum turbine entry temperature of 1430 K at steady-state operation is not exceeded. Even during acceleration no appreciable change in the temperature of the hot parts occurs. However, it must be admitted that this takes for granted optimal matching of the compressor, whose surge margin must be matched to the position of the operating line corresponding to TET = const. (The operating line is constant for steady-state and transient conditions).

From the point of view of pilot safety the shortening of the acceleration time of the gas generator is of particular importance for a twin-engined attack helicopter with regard to the possible failure of one engine at low altitude.

6. Design

In order to adapt the reference engine to operation with heat exchanger, the following design modifications in addition to the dimensioning carried out for comparison purposes were required (Fig. 10):

- New compressor outlet casing with four connexions for ducting air from the compressor to the heat exchanger.
- New combustion chamber casing with four tubes for ducting hot air from the heat exchanger to the combustion chamber.

- New power turbine casing required by the introduction of variable vanes for the two-stage power turbine.
- New gas generator turbine bearing housing.
- Slightly modified power-turbine outlet casing.
- Slightly modified power-turbine rotor involving the incorporation of a rotating spacer between the two impellers of the power turbine.

The profile heat exchanger used is of V-configuration and is suspended in such a manner as to permit movements caused by thermal expansion in all directions. The supply ducts (4 in each case) between the compressor and heat exchanger and between the heat exchanger and combustion chamber are provided with bellows to compensate for relative movements between the turbine and heat exchanger casing.

Both engines were fitted with an infrared suppressor to fulfil the following functions:

- To block the view of hot engine parts,
- To cool the visible outer exhaust casing, and
- To mix cold air with the exhaust in order to lower the exhaust gas temperature to an acceptable value.

Accordingly, the infrared suppressor [1] in question consists of an ejector and a mixer, with the ejector flanged to the exhaust or heat-exchanger casing and the mixer secured to the airframe. The ejector has four distributor nozzles for blowing the engine exhaust gas into the four larger-area ducts of the mixer simultaneously mixing it with cold air. Efficient cooling of the partly visible outer sections of the mixer offers the advantage to manufacture the mixer in aluminium for low weight.

A comparison of the length of the two engines reveals an approximately 400 mm greater length of the engine with heat exchanger. However, as far as the maximum cross sections for the engine cowling are concerned, the comparison shows that when infrared suppressors of the same design are used for both engines, roughly the same cross-sectional dimensions for the installation space and the cowlings are required.

7. Overall weight of the power system

The absolute fuel consumption as a function of power of the regenerative and reference engine is shown in Fig. 11 . In comparison with the reference engine the regenerative engine shows an improvement in fuel consumption of approximately 27% at medium power.

Fig. 12 gives a tabulated summary of the overall weight of the reference engine and the regenerative engine. The approximately 28 kg heavier turbomachinery section of the engine with heat exchanger is attributable to the following differences from the reference engine:

- Lower specific power despite higher turbine entry temperature, because of greater pressure losses (heat-exchanger matrix, ducts to and from the air and gas sides).
- Lower efficiency of the variable power turbine.
- Actuation system weight of the variable guide vanes of the power turbine.
- Somewhat lighter infrared suppressor, because of the lower exhaust temperature.

Including all the parts making up the heat-exchanger assembly, the overall weight of the power system comes out to be 552 kg, which means a weight saving of 35 kg in comparison with the reference engine.

8. Life-cycle costs

The assumptions made when estimating the life-cycle costs of the engines of a fleet of helicopters are summarized in Fig. 13 . Comparison of the life-cycle costs was made under the assumption of the same flight performance of the helicopters. The lower take-off weight in the case of the regenerative engine leads to a reduction of the installed power and, thus, to an even greater reduction in the overall power system weight than determined in section 7. Consequently, a total reduction in the overall weight of the power system of 122 kg is attained with the twin-engined attack helicopter. Assuming a similar saving in the airframe weight (including the rotor), this leads to a reduction in the take-off weight of 244 kg (5.4%).

According to Fig. 14 (left-hand half), for 250 flying hours per year and distribution of development and product support costs over 1000 engines, this gives an increase in the life-cycle costs of 4.4% against the reference engine at the present fuel prices. Because of the considerably smaller quantity of fuel per mission required by the engine with heat exchanger an increase in the fuel price of more than roughly 60% leads to lower life-cycle costs of this engine in comparison with the reference engine. Based on 500 flying hours per year, life-cycle costs of the regenerative engine are always lower for the fuel price range under consideration.

The right-hand half of Fig.14 further shows the influence of the development and product support costs on the life-cycle costs with distribution over a greater number of engines. For 250 flying hours per year and at current fuel prices, an advantage with regard to the life-cycle costs is attained for the engine with heat exchanger assuming a production figure of 2000 engines.

9. Development problems

Decisive for the successful development of a regenerative engine of this category is the effectiveness of the heat exchanger and its thermal and mechanical behaviour. For the reasons given in section 3, the heat-exchanger concept selected is particularly suited for the operating conditions in this case. Further, the design principle, similar to that of a tube-type heat exchanger, and the detailed manufacturing and test experience with tube-type heat exchangers under comparable operating conditions, give good reason to expect a positive course of development. Special emphasis in this context is to be paid to a functional design, which at the same time promises minimal production costs.

As far as the combustion-chamber section is concerned, the hot air coming from the heat exchanger at temperatures of up to 900 K signifies considerably greater thermal loading of the flame tube than would be encountered in conventional engines of this class. This circumstance calls for an accurate flame-tube structure with minimal expansion of the flame-tube surface. Because of the very high thermal loading even the injection nozzles will probably have to be cooled, meaning that the fuel will have to be drained off after shutdown of the engine.

The variable stators of the two-stage power turbine call for a high degree of variability with minimal radial clearances of the vanes. Here it will be appropriate to build on MTU's experience with a single-stage variable turbine. In the case of the two-stage variable turbine it will be necessary to guarantee easy, but simultaneous low-play operation with a very high number of movements in particular.

Finally, the regenerative engine requires a control unit which permits the performance potential of the engine under transient and steady-state conditions to be fully utilized. Because of the allocation of service data, described in section 2, it will in this case be necessary to combine a digital control unit with quick-reacting sensors in such a way as to provide for a favourable-cost and reliable control unit design.

10. Conclusions and evaluation of results

From the above mentioned it becomes apparent that conventionally designed modern turboshaft engines of the power class under discussion here serve very well as the starting point for developing an engine with heat exchanger. Admittedly, this is true only when the mean power of the engine is relatively low, as it is in the case of the mission of an attack or transport helicopter for example.

It may be concluded from the determination of the engine life-cycle costs that because of higher development and production costs turboshaft engines with heat exchanger are the more advantageous, the greater the number of marketable engines of a single model and the higher the annual utilization for the same service period. From the economic point of view, therefore, at least for the time being, engines with heat exchanger with an assumed service life of 3750 hours are not attractive. However, in keeping with new requirements, the attack and transport helicopters presuppose an overall running time of at least 6000 hours, corresponding to a service period of 24 years. This means that the reduced fuel consumption of the regenerative engine would make itself much more felt. This applies all the more when one considers that we shall have to reckon with an appreciable increase in the fuel costs in the long run. Over and above this, the findings show that turboshaft engines with heat exchanger promise interesting advantages for helicopters and aircraft built in greater numbers and operated over longer periods.

The very good acceleration capability of the engine with heat exchanger contributing significantly to safety in the event of failure of one engine, is of interest as far as the twin-engine attack helicopter flown at low altitudes is concerned.

There are no obvious advantages of the regenerative engine over the reference engine as far as infrared radiation is concerned.

With regard to the installation conditions, despite its somewhat greater length, the engine with heat exchanger is not expected to present any serious disadvantages, especially as both engines have practically the same transverse dimensions.

Acknowledgements

The authors are particularly indebted to the MTU München GmbH and to the Federal German Ministry of Defence for permission to publish the present study.

The authors also wish to thank Mr. A. Rohra, who carried out the design work.

Reference

[1] B. Barlow and A. Petach
"Advanced Design Infrared Suppressor for Turboshaft Engines"
Presented at the 33rd Annual National Forum of the American Helicopter Society, Washington, D. C., May 1977
Preprint No. 77.33-73

		Attack Helicopter	Transport Helicopter
Max. Take Off Weight	T	4,5	6,0
Number of Engines		2	
Engine Power (Design Point)	kW	900	
Mean Power	%	46	55
Engine Life	Hrs	6000	
Utilization	Hrs/Yr	250	300
Mission Endurance		2 Hrs. 30 Min. + 20 Min. Reserve	
Start/Stop Cycles per Mission		1	2

Fig. 1 Helicopter Mission Essentials

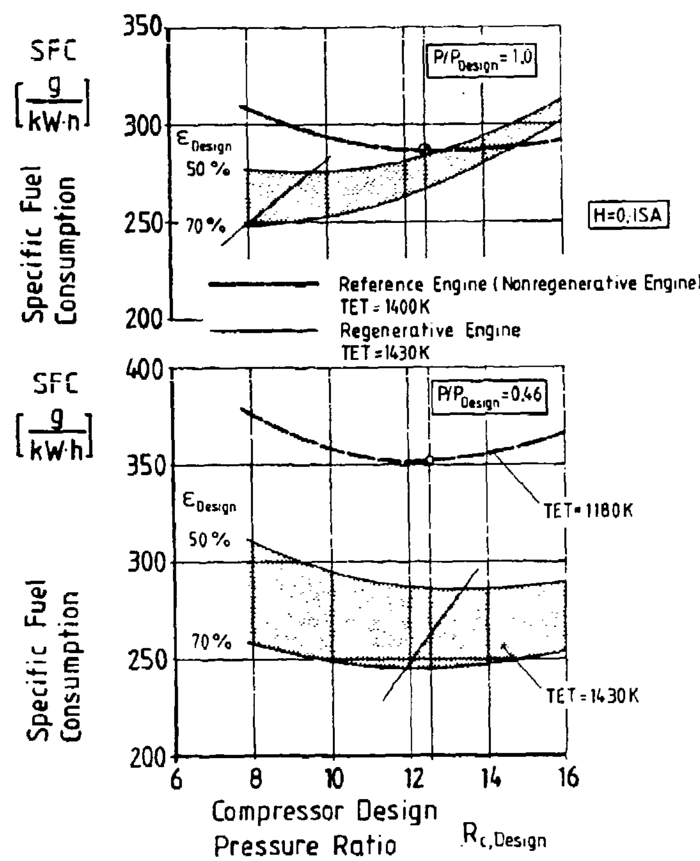


Fig. 2 Influence of Design Pressure Ratio on SFC at Maximum and Part Power

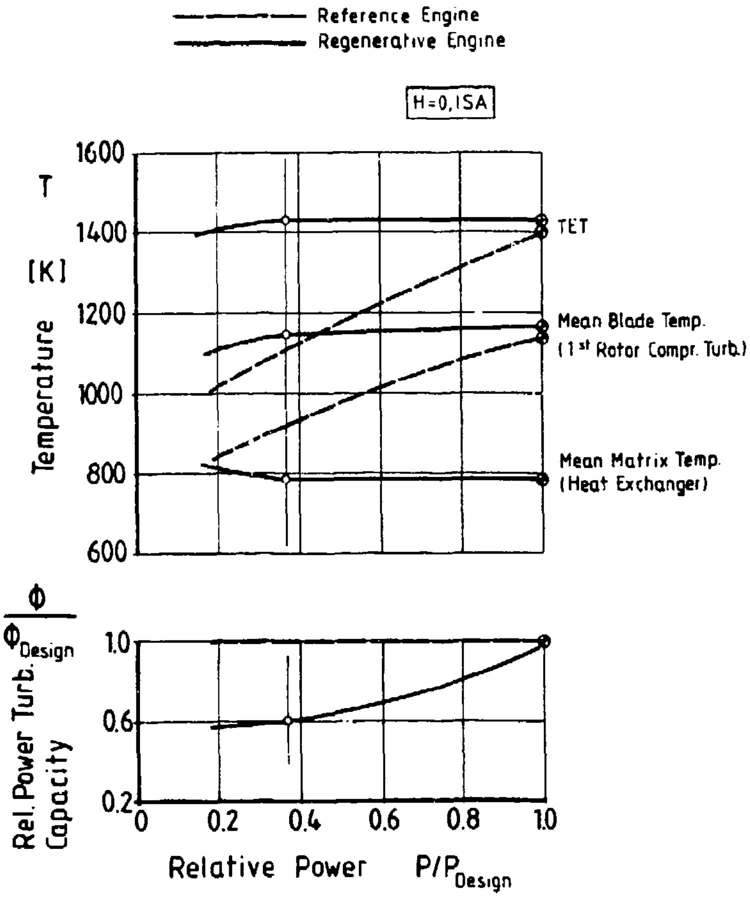


Fig. 3 Influence of Power Turbine Flow Capacity Control on Important Temperatures

Reference Engine:

TET = 1400 K
R_C = 12,5

			Reference Engine	Regenerative Engine
Total Engine Lifetime	L	Hrs	6000	6000
Number of Missions			2500	2500
Start/Stop Cycles			2500	2500
Number of Full Thermal Cycles			15000	2500
Trend of Rotor Blade Temperature (Compr. Turbine)	T_{BLADE}	K	= f (P)	Constant (Isothermic Concept)
Temperature Penalty (Creep Life)	ΔTET	K	-	- 40
Temperature Bonus (Cyclic Loading)	ΔTET	K	-	+ 70
Resulting Temperature Difference	Σ(ΔTET)	K	-	+ 30

Fig. 4 Influence of Creep and Cyclic Loading on Achievable TET

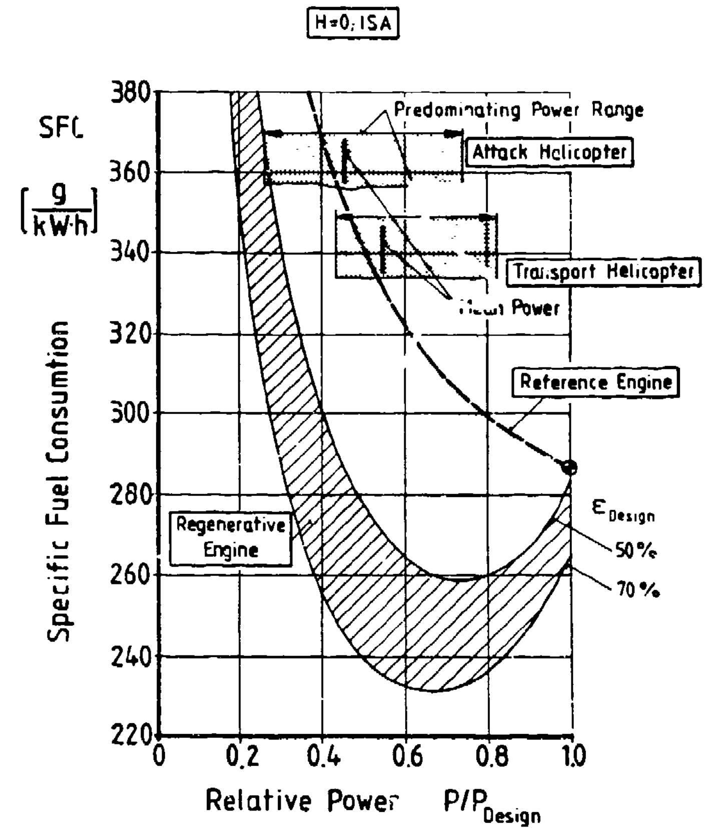
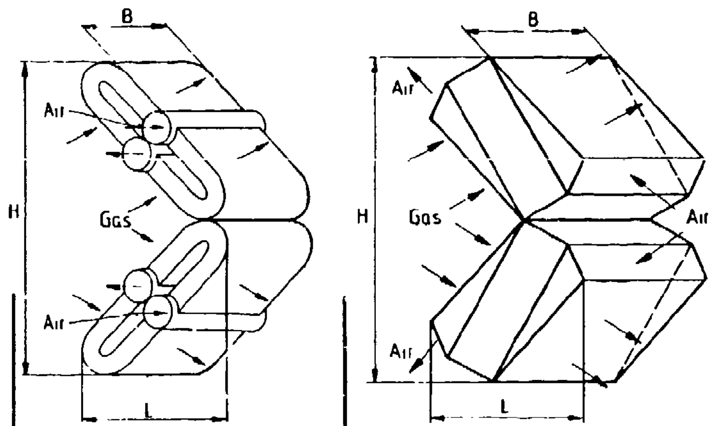


Fig. 5 SFC at Part Power



Type of Recuperator	Tubular	Profile	Plate-Fin
Flow Arrangement	2-Pass Cross-Counterflow		Counterflow
Hydraulic Diameter Air/Gas Side mm	3,0/2,7	2,2/1,6	3,0/1,5

Fig. 6 General Arrangement and Flow Pattern of Heat Exchangers Considered

TCT = 1430 K
R_C = 12,5
P_{DESIGN} = 900 kW

Recuperator Effectiveness:	60 %
Total Pressure Loss :	7 %

Type of Recuperator		Tubular	Profile	Plate-Fin
Matrix Volume $V_{MATRIX, NET}$ $V_{MATRIX, GROSS}$	 m^3 m^3	 0,030 0,038	 0,017 0,023	 0,026 0,040
Matrix Weight * $W_{MATRIX, NET}$	 kg	 23	 24	 23
Front Area H x B	m^2	0,24	0,18	0,26
Length L	m	0,69	0,36	0,30

* without bracing filler

Fig. 7 Matrix Data of Heat Exchangers Considered

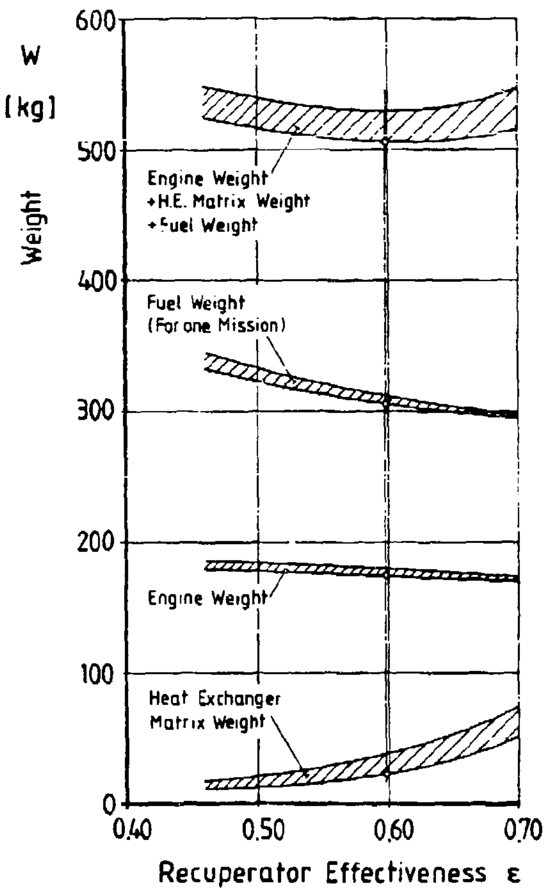


Fig. 8 Heat Exchanger Optimization

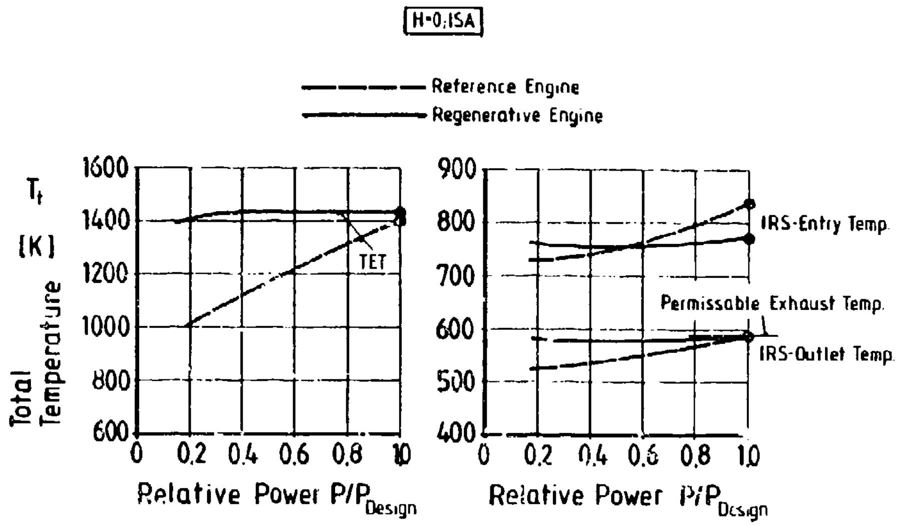


Fig. 9 Comparison of Gas Temperatures at Entry and Exit of IR-Suppressor

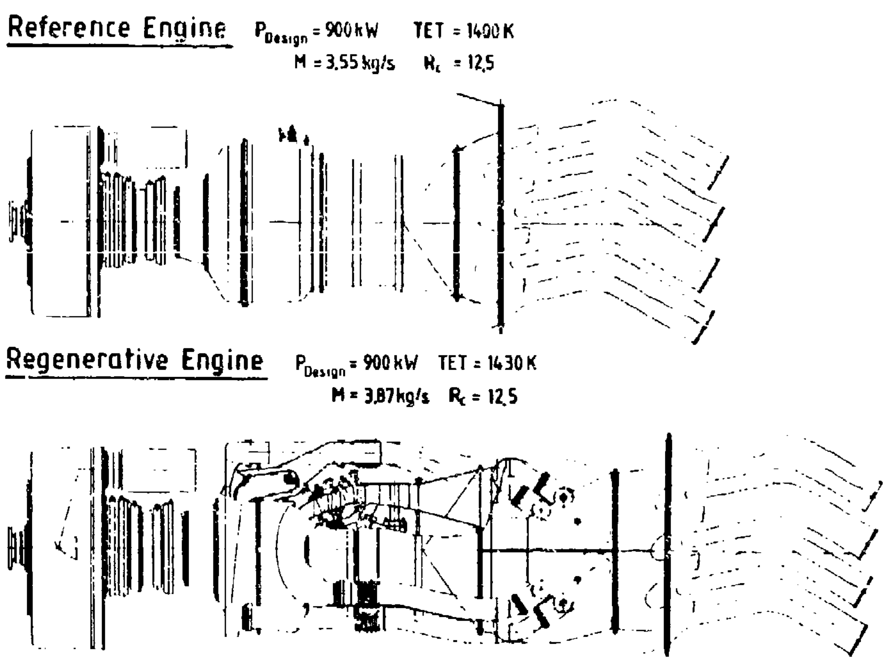


Fig. 10 General Arrangement of Engines Considered

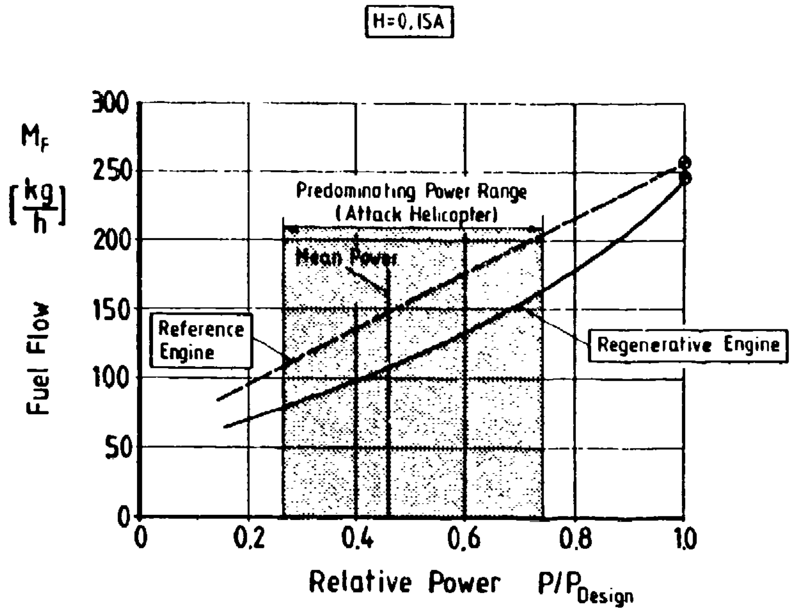


Fig. 11 Comparison of Fuel Consumption of Engines Considered

P_{DESIGN} = 900 kW

		Reference Engine	Regenerative Engine
Mission Fuel Weight	kg	417	306
Engine Weight	kg	148	176 (without Recuperator)
IR-Suppressor Weight	kg	22	17
Recuperator Gross Weight	kg	-	53
Total Weight	kg	587	552

Recuperator (53 kg): Matrix 25 kg; Supply Ducts 28 kg

(The Weights refer to 1 Engine)

Fig. 12 Comparison of Total Power System Weight

Number of Helicopters (Twin Engined)	200
Service Time Period	15 Yrs
Utilization	250 Hrs/Yr
Engine Produced	1000
Economic Condition	Jan. 80
Fuel Price	0,65 DM/kg

Helicopter	
Mission Equipment	Same
Payload	Same
Airframe Cost (Without Equipment)	1100 DM/kg

	Reference Engine	Regenerative Engine
Max. Take Off Weight	4500 kg	4256 kg (- 5,4 %)
Engine Design Power	900 kW	851 kW (- 5,4 %)
Engine Weight	170 kg	231 kg (+ 35,9 %)
Mission Fuel Weight (Per Engine)	417 kg	295 kg (- 29,3 %)
Development Cost	Base	+ 23 %
Aquisition Cost	Base	+ 24 %
Product Support Cost	Base	+ 16 %
Maintenance Cost	Base	+ 16 %

Fig. 1: Assumptions for LCC Analysis

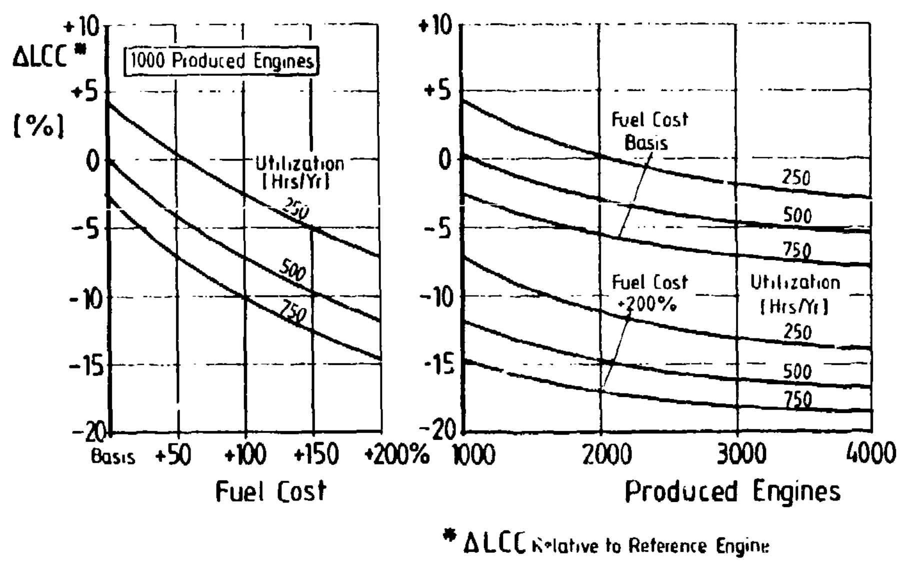


Fig. 14 Comparison of LCC for Engines Considered

DISCUSSION

H.Saravanamutto, Ca
Have you considered the expected high temperature of the power turbine at part load?

Author's Reply
This was considered with the turbine temperature being reduced at loads below 37 percent of maximum power.

H.Saravanamutto, Ca
Did you consider the deceleration response in your design study?

Author's Reply
No deceleration problem is anticipated because of the nearly constant heat exchanger mean matrix temperature at part load with subsequently constant overall content in heat energy (see Figure 3).

M.Hudson, US
Did you consider the rotary heat exchanger in your design and could you tell us why it was rejected?

Author's Reply
It was rejected because of the high power level and the high engine pressure ratio as well as potential integration problems.

W.Schneider, US
In reducing the power to 40 percent at constant cycle temperature, have you considered the compressor surge problem?

Author's Reply
Yes, the compressor would have to be designed with an appropriate surge characteristic.

K.Rosen, US
Because of potential drag penalties, did you parametrically study the effect of increase in drag on increased fuel consumption?

Author's Reply
No, the aircraft geometry remained roughly the same for both engines because of their nearly identical maximum cross sectional dimensions.

Unknown Questioner
Could you comment more on the differences in weight between the two engines?

Author's Reply
As shown in the paper, when considering the reduced mission fuel weight for the regenerative engine, the total weight is actually less than that of the reference engine (see Figure 12).

LUBRICATION BREAKDOWN BETWEEN GEAR TEETH

by

B.A. Shotter - Chief Design Engineer (Gears)
Westland Helicopters Ltd., England

Damage to gear teeth attributable to "lubrication failure" is an important failure mode. Though much has been written concerning this problem, several important aspects of the damage initiation have received little attention. Detailed observations of the early stages of breakdown suggest that one may not be dealing with a single process· hence great care is necessary when drawing conclusions from observing a damaged pair of gears.

A study of a number of these critical areas and discussion about others, highlights the complexity of the problem. It is only by fully understanding the nature of the problem, that satisfactory remedies can be found.

1. Introduction

Since many thousands of words have already been devoted to this subject one might justifiably question the need for a further paper with a general title of this nature. The examination of failed gears from many fields of operation, not only aeronautical, but also industrial, automotive and marine, have shown characteristics which suggest that there are several initiating mechanisms involved. The subsequent damage may develop along similar lines and this can obscure the real cause of the problem unless the early symptoms are observed and understood. This paper hopes to describe some of the various initiation processes and to improve the general understanding of the problem.

2. Basic Observations of Failed Components

2.1. Position of early stages of damage

The sequence of events leading up to major damage is always important in trying to prevent such a problem recurring. However, there are inevitably some cases where the damage has progressed so far, that it is no longer possible to trace back the failure to its true origin. The increasing interest being shown in health monitoring techniques could well lead to an earlier awareness of damage and an improved ability to find the damage initiation areas. (Ref. 1)

When looking at various examples of lubrication breakdown a lot of the cases can have their origins traced back to the tooth edges; either the ends or the tips. One example of this is shown in Figure 1. Here the tip of a spiral bevel gear is showing markings extending down the flank. Whilst the photograph shows these markings quite clearly, they were hardly visible in the high level of illumination by fluorescent tubes frequently found in workshops. It was only by using a beam of light at a very shallow angle to the surface that these markings became so obvious.

Other examples are found where the early damage is clearly not associated with the tooth edges. Here it is important to consider the implications of other factors such as surface finish or surface treatments. Sometimes breakdown can be seen to extend away from some relatively small surface defect such as a small pit or an indentation perhaps caused by a piece of debris passing between the teeth.

The significance of these relatively minor markings in the early stages of damage development may be modified considerably by the nature of the gear materials, the presence of common factors between the tooth numbers of meshing gears and the character of the speed and load spectra.

2.2. Influence of time on initiation

Whilst physical appearance may show the regions where damage has been initiated, it is also important to know when it started. Probably the commonest time of initiation is during the early life of a gear set, when the surfaces are still changing during the somewhat ill-defined process of "running-in". Certainly during this time, temperatures will tend to increase and oil viscosity will fall, rendering the generation of hydrodynamic oil films less efficient. However, the process leading to breakdown is not solely related to the progressive loss of oil film thickness until intimate metallic contact is established. Frequently there appears to be a sudden collapse to failure, though the oil film thickness at which this happens can be quite variable. (Ref. 2, 3)

Apart from early life failures there are many examples of lubrication breakdown after prolonged periods of running. Some of these can be associated with an interruption in lubricant supply; usually it is possible to find evidence of heating if this has occurred. Not infrequently one finds new oil added and the gears allowed to run on, thus polishing out the evidence of heating on the operating flanks. But careful examination of the edges of the contact area will often still show the tell-tale temper colours that demonstrate that high temperatures have been reached.

Another form of long life lubrication breakdown can be associated with fatigue of the metal surfaces. This has been demonstrated in the laboratory where sudden scuffing has occurred on surfaces which were well past any running-in phase (Ref. 4). Sometimes on gears it is possible to see the presence of fine micro-pitting (Fig. 2) on some areas of the teeth where the scuffing damage has not obliterated all the earlier fatigue damage. However, it is also possible for a scuffed surface to start to fail by fatigue, so it is important to find out which came first! With carburised and case hardened gears it seems that pitting after scuffing is most commonly associated with the relatively lightly damaged areas near the edges of the scuffed zones.

In considering when a particular failure occurred it is important to distinguish between time and life. A gear which has had a "life" of several thousand hours could suffer lubrication breakdown after a gearbox overhaul. Even minor disturbances to gear alignment, perhaps due to a bearing change, can initiate a failure. This is particularly common when there is evidence of wear on the teeth. Slight displacement of the contact area can cause excessive local loading at an edge which can then trigger off the failure. Other gears have failed relatively soon after standing idle for a long period. This may be associated with corrosion damage. Some scuffed gears that have been examined have shown typical corrosion marks on their non-working surfaces although the flanks were too damaged to positively identify any corrosion as the initiation site.

2.3. Additional Observations

Whilst most failures are associated with the contact between one set of driving tooth surfaces and a corresponding driven set only. It is important to relate these together correctly. For instance reassembly of two gears both having even numbers of teeth can result in tooth surfaces contacting that have never previously run together. Any gear pairs having a common factor in their tooth numbers can suffer in this way.

Whilst load, or more specifically, high local loading, can be responsible for initiating damage, this load can be self generated. Gears operating with very low backlash can start to contact on both flanks. This can generate more heat and cause the gears to expand thus imposing even more load on the teeth. The combined effect of load and temperature in these circumstances will frequently cause lubrication breakdown. Thus careful observation of the nominally unloaded surfaces may provide useful evidence to explain a failure.

It is still possible for gears to pull themselves together, even when relatively large backlash exists. This can occur with gears having virtually all-addendum driving teeth, the explanation of how it can happen is given in Appendix I. However, several cases of tooth damage have been observed where the only explanation seems to have been this unusual tendency for the gear teeth to pull into mesh with one another; improbable though this sounds at first.

3. Contact Conditions

3.1. General Tooth Contact Geometry

The instantaneous contact between conjugate profile gear teeth will be a line. For Spur gear pairs this is always a straight line parallel to the gear axes. For helical gears this line may be curved. For Involute helicals the line is again straight but is now inclined through the plane containing the shaft axes. Gear tooth contact stresses are defined relative to a plane normal to this contact line. What is often forgotten is that the area of one tooth is relatively small and that the line of contact has to stop somewhere! (Fig. 3). These discontinuities can have quite a considerably effect on the local stresses at the end of the contact line, even when the basic tooth alignment is perfect. In the presence of manufacturing errors the peak stress near the end of the contact line can be very much higher than the theoretically calculated value for the basic tooth geometry.

3.2. Modified Contact Geometry

The fundamental behaviour of a line contact of finite length has been studied in connection with roller bearings (Ref. 5). To alleviate these adverse end stress conditions special end profiles are often used together with a slight barrelling of the cylindrical form. The Involute tooth system makes some attempt to provide a modification to the edge conditions by introducing "tip" or "tip and root" relief. Though the application of these is often rather crude in concept.

Since many lubricant breakdowns are associated with the tip contact conditions of either the pinion or the wheel, the influence of tip relief on such failures could be quite significant (Ref. 6). Whilst it is sometimes recommended that the ammount of tip relief should just equal the elastic deflection of the tooth, experience has shown that rather more relief is usually necessary to prevent the contact line "falling off" the edge of the tooth. Whilst some improvement is found with increasing relief, major benefits seem to acrue from the total containment of the contact line within the flank area. Such behaviour seems difficult to account for on the basis of contact stress alone and the section on lubrication contributes to the understanding of this observation.

3.3. Edge Geometry

Whilst tip relief is usually applied during the normal manufacturing process producing the tooth flanks, the boundaries of these surfaces to the tooth ends and tip normally receive a separate dressing operation. The character of this operation and the nature of the finish produced can significantly modify the failure characteristics of the gears. Thus,although such operations are frequently felt to be relatively unimportant,they can become critical to the satisfactory performance of some gears.

3.4. Surface Properties

If the surface finish on the boundary edges is important, what about the tooth flank itself? This question cannot be answered in isolation. The optimisation of surface finish on gear teeth is a problem having many facets and the answers seem to vary in different cases. To try to define the characteristics of an optimum surface is itself a difficult problem. In consequence all that is done here is to note some factors of significance, but not to draw any hard and fast conclusions from them.

i) A surface finish "lay" running across the sliding direction can be beneficial in interrupting the mini-welds before they develop into larger scale damage. There are also benefits if the "lay" runs across the rolling motion direction since this appears to improve the lubrication as well (Ref. 7).

ii) Surface finish amplitude whilst often not appearing to relate directly to lubrication breakdown, can influence behaviour through its effect on operating temperature. It has been shown that high initial surface roughness can result in higher coefficients of friction on run-in surfaces (Ref. 8.). Thus if increasing temperatures render failures more likely, as will happen with lubricants operating purely on a viscosity basis, then smoother initial finishes will be beneficial. However, many oil additives become more reactive with increasing temperatures and thus the increased operating temperatures resulting from the use of rougher surfaces can have beneficial effects in some circumstances.

iii) The presence of occasional deeper scratches on a surface can have a disruptive effect on the oil film generation. Some cases have been seen where the surfaces adjacent to a scratch seemed almost on the point of scuffing. Had such an event occurred, it would almost certainly have obliterated the scratch and the true origin might never have been found.

iv) "Anti-Scuffing" treatments can have conflicting effects on the surfaces. For example a phosphate treatment produces an "anti-scuff" film, but at the same time tends to increase the surface roughness of the metal. Cases have been seen where the protective film appears to have worn off with time and then scuffing has occurred, but only within the more heavily worn area. Other gears of the same design and operating under similar loading have not failed when the depth of surface etching was less.

Another problem encountered with a phosphate "anti-scuffing" treatment concerned an edge breakdown. In this case it was shown that the "protective" treatment actually made the failure worse. So that, although these treatments may prove satisfactory in some circumstances, they cannot be considered as a universal panacea whenever lubrication breakdowns occur.

4. Lubrication

4.1. The Lubricant

Although it might seem to be a fairly probable cause of lubrication breakdown, the lubricant itself is only one link in the load carrying chain, and major improvements in load carrying performance can usually be made without resorting to oil changes. As mentioned in 3.4. (ii) the two main factors of a lubricant are its viscosity characteristics and its chemical behaviour. Higher operating temperatures tend to make one more dependent on the chemical characteristics, although higher speeds need oils of lower viscosity so these two aspects are not necessarily in conflict. However, Macpherson has shown (Ref. 9) that the scuffing load of given oils can vary significantly with temperature, the most critical region being in the 60 - 90°C region. This seemed to demonstrate an inadequacy in the development process of the oils, since operation at these temperatures is almost inevitable during some periods in a units life.

Whilst the fundamental properties of the oils are important, the selection of an oil for a given gearbox duty will depend on the operating characteristics of the gears within it. Where a single gear pair is concerned, a viscosity can usually be chosen to suit the pitch line velocity of the teeth. When several reduction stages are contained in the same gearbox the problem becomes more difficult, since the high torque low speed output stage would normally prefer a much higher viscosity than a low torque high speed input stage. Thus compromises are necessary and they can show up in any lubrication breakdown effects that are noted. With Military equipment, the logistics demand of keeping a minimum number of oil types can also affect the selection of a lubricant.

4.2. Basic Elastohydrodynamic Lubrication (e.h.l.)

Since the majority of gears in use are Involute the argument will be developed around the contact conditions for the basic Involute gearing system although the concepts can be adapted to other tooth systems. The instantaneous line of contact on spur gears will sweep from root to tip on a driving gear or tip to root on the driven member. As the contact moves over the oil wetted surface, an oil film becomes trapped between the surfaces due to the high viscosity that develops at the contact pressures existing between the teeth. Such conditions have been investigated using disc machines which can maintain continuously the conditions at one point during the action of two teeth (Ref. 10). The problem with gear teeth is that the e.h.l. contact has to start and finish on every tooth.

Let us first consider the starting process. The tip of the driven tooth will be the first point to be contacted, so one can observe the distance between the corner of this tooth and the lower flank of the driving gear. As the distance reduces there is no basic reason why a significant oil film should develop. It is only when a contact pressure zone starts to sweep over the surface that the basic requisites for generating the oil film can be found. Thus the initial contact has to tolerate the high sliding conditions with an inferior lubrication (Fig. 4a). The benefits of "tip-relief" in reducing load during this phase are, therefore, likely to interact beneficially with the e.h.l. development process.

Looking now to the way in which the contact ceases at the tip of the driving gear, the geometric conditions are reversed but from the e.h.l. point of view the action is significantly different. The high contact pressure zone exists already and is moving towards the tooth edge. The film thickness is generated at the ingoing edge of the Hertzian zone and the radii of curvature of these approaching surfaces together with their velocities towards the conjunction will influence the magnitude of the film. When the pressure zone becomes very close to the tip the effective radius of one surface suddenly becomes very small and causes an inferior lubricant film, this again at a point where high sliding velocities exist (Fig. 4b). It is not surprising that scuffing can start in such areas.

The behaviour at surface discontinuities can also be seen where a relatively deep scratch crosses a surface. Since the scratch relieves the pressure in the oil film development process, it acts in a similar way to the tooth edges. The amount of surface interaction adjacent to the scratch edges is much greater than on the rest of the surface, demonstrating the inferior oil films at the sudden discontinuity.

With teeth having a helical form the instantaneous contact will exist as a diagonal line across the tooth flank. Thus the two ends of the contact line coexist with the start and finish conditions already described for spur gears. Although the central portion of the contact zone can be generating near perfect oil films, the edge discontinuities are still the regions where lubrication breakdowns most frequently occur. Similar behaviour is often observed on spiral bevel gears.

4.3. E.h.l. at the ends of Helical Gear Contact Lines

Whilst the previous section referred to the lubrication conditions at these points, the situation is rather more complex than suggested there. The oil film thickness in an e.h.l. contact is largely determined by the conditions at the entry to the contact area. For a line contact this is fairly simple to define. The entraining velocities of the surfaces normal to the contact line and the radii of curvature of the surfaces in the same directions are relatively basic properties and easily established (Fig. 5). What about the ends of the contact line though?

Firstly the contact at the tip of the driven member with the root end of the flank on the driving tooth. When the first contact between these teeth occurs it is only a point right at the end of the teeth. What velocity is to be used under these circumstances? As far as this end of the contact line is concerned it is sweeping across the facewidth with a velocity far faster than the sweep velocity of the subsequently developed line up the tooth. The effective radii of curvature in this direction are also larger. Thus this end of the contact line can operate under conditions attempting to improve the local oil film thickness.

At the other end of the contact line, however, the conditions are very different. The sweep velocity of this end across the tooth is virtually the same as the other end of the contact line, but here this cannot help to generate an oil film. At this end the film thickness is predetermined by the entry conditions of the line contact and, as already shown, these tend to be less effective at this point.

Diagramatically the relative film generation is shown in Figure 6. Thus even when reasonable attention is given to the load discontinuity effects at the ends of the contact line, there still exists an imbalance in the scuffing tendencies at these points. Certainly damage seems to start at the exit end rather more frequently than the entry end.

4.4. Lubricant Supply

Whilst tooth geometry, torque loading and speed all affect the onset of failure in a relatively predictable manner, the provision of an optimised lubricant distribution over the tooth surface is far more uncertain. The lubricant serves two distinctly separate functions. It produces the film separating the metallic surfaces, but it also serves to transfer thermal energy from the gears. In many cases the majority of the oil is required for the second function. However, excessive quantities of lubricant can generate heat as "churning losses". These increase rapidly with speed and it is usually found that dip lubrication becomes unsatisfactory above 20 m/s pitch line speed unless special steps are taken. Lubrication breakdowns can be aggravated either by excessive temperature in the presence of adequate quantity of lubricant or by an inadequate supply of lubricant which can lead to higher temperatures. Distinguishing between these is not always easy. The temperature distribution on the gear case can sometimes be helpful.

One example where the wrong conclusions were made involved a low ratio bevel gearbox. The gears were relatively small in large casing and sliding speeds were low. A reasonably low pitch line speed was consistent with the dip lubrication used. Because of the large casing, oil picked up by the dipping gear was thrown to the casing walls and under cold conditions the gear mesh could be starved of oil. Assuming that the oil properties were inadequate for the duty, an even higher viscosity oil was used which made matters worse! Finally the gears were given a baked-on Molybdenum Disulphide treatment, which cured the immediate failure symptoms, but damage still occurred later in the equipment life. Although excessive space was fundamentally the cause of the problem in this case, more problems are usually caused by a lack of space. Uniform clearances around the tip circles may look nice on a drawing board, but they can make it difficult for oil to get away from a gear rim and thus cause increased churning losses.

With higher speed gears the lubrication is usually by means of jets. The position of failure initiation relative to the jet can be significant; on larger facewidth gears, twin jets applying the same oil quantity can be successful where a single jet system has failed. However, the windage around the high speed gears can break-up an oil jet and although it may be pointing in a certain direction this is no guarantee that it will strike the gear in that position. Sometimes local wind deflectors can be beneficial in preventing such disruption.

Often the oil velocity leaving a jet is lower than the pitchline veloc y of the teeth. In such cases it can seem improbable that the oil would ever reach the ro s of the teeth (Ref. 11). However, contact with one gear can accelerate the oil and thereby enable it to enter the tooth space of the opposing gear rather more easily than might be expected.

Frequently oil droplets are ricochetting between the casing and the gear many times; each cycle transferring a little thermal energy to the casing. Such a process is the explanation of why sometimes it is found better to direct oil jets on the outgoing side of the mesh. When directed to the inlet side it is possible for the tooth action to move the oil to the side of the gearbox and not to generate as effective a spray to promote the thermal interchange.

Lubrication failures can sometimes be initiated during the starting cycle. If the equipment has stood for some time, very little oil will be left on the teeth. Should the start-up acceleration be very high, it is quite possible for high torques to be developed before the gears have made a complete revolution. Thus some teeth may experience almost unlubricated operation. Subsequent revolutions in the lubricated state may not be able to prevent the damage growing since, in the cold state, the chemical activity of the additives can be very low.

As can be seen there are many possibilities for the lubricant supply to influence the operating properties of the teeth. When problems arise there are usually a lot of possible alternatives. A careful analysis of the evidence can often help to cure the trouble with the minimum amount of change to the system.

5. Conclusions

Whilst criteria exist for assessing some potential lubricant failure conditions, there are many other significant factors. Because of this it is questionable whether these calculations serve a really useful function. Apparant limits found in one application can be significantly exceeded in others. It is important to recognise that there are a lot of possible initiation mechanisms and not to attribute all lubrication breakdowns to the same cause.

Improved reliability will only be achieved by recognising the different factors that can cause failures and ensuring that all these factors are given due consideration. When striving to improve the load capacity of gear systems it is inevitable that failures will occur. Only if the true origin of a failure is recognised will it be possible to successfully modify future designs so that higher loads can be achieved.

6. Acknowledgements

The Author wishes to thank Westland Helicopters Ltd. for permission to publish this paper. He also acknowledges the help of many people in the investigation of gear tooth problems which provided the background for this paper.

APPENDIX 1

EFFECT OF FRICTION AT THE TOOTH CONTACT ON THE TOOTH NORMAL FORCES AND SEPARATING FORCES

Consider initially an elemental tooth flank surface under static or zero friction conditions as shown in Figure A1 (a). A given tangential force L_T is required from the gear. To produce this, a normal load L_N will have to act on the surface which is inclined at a pressure angle α . This will also cause a separating force L_S acting radially.

$$\text{Thus} \quad L_N = \frac{L_T}{\cos\alpha} \quad \text{or} \quad \frac{L_N}{L_T} = \frac{1}{\cos\alpha} \tag{1}$$

$$\text{and} \quad L_S = L_T \tan\alpha \quad \text{or} \quad \frac{L_S}{L_T} = \tan\alpha \tag{2}$$

If now one introduces a frictional effect at the contact as in Figure A1 (b) due to the upward movement of the contact point, then components L_{TN} and L_{SN} will result from the Normal load as in the previous case. But a frictional force L_F will exist in the plane of the surface where

$$L_F = \mu L_N$$

This can again be resolved into tangential and radial directions to give:-

$$L_{TF} = L_F \sin\alpha$$

$$\text{and} \quad L_{SF} = L_F \cos\alpha$$

Hence, the required tangential force L_T will be given by:-

$$L_T = L_{TN} + L_{TF} = L_N \cos\alpha + \mu L_N \sin\alpha$$
$$= L_N (\cos\alpha + \mu \sin\alpha)$$

$$\text{or} \quad \frac{L_N}{L_T} = \frac{1}{\cos\alpha + \mu \sin\alpha} \tag{3}$$

The resultant separating force L_S will be given by:-

$$L_S = L_{SN} - L_{SF} = L_N \sin\alpha - \mu L_N \cos\alpha$$
$$= L_N (\sin\alpha - \mu \cos\alpha)$$

$$\frac{L_S}{L_N} = \sin\alpha - \mu \cos\alpha \tag{4}$$

But relative to the tangential load L_T one has to combine (3) and (4) to obtain:-

$$\frac{L_S}{L_T} = \frac{\sin\alpha - \mu \cos\alpha}{\cos\alpha + \mu \sin\alpha} \tag{5}$$

The results of equations (3) and (5) are plotted in Figure A2. The first graph shows how the increase in friction tends to reduce the normal tooth loading. The constant friction curves show minima which for typical lubricated friction co-efficients will be below most working pressure angles. Hence, increased pressure angles will cause increasing normal tooth pressures.

The separating load curves show intercepts with the zero axis at values corresponding to the minima of the normal load curves.

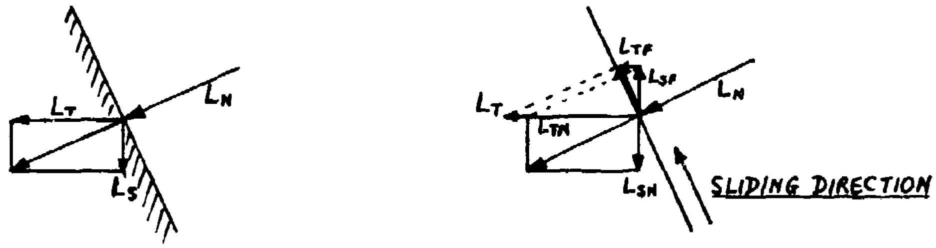
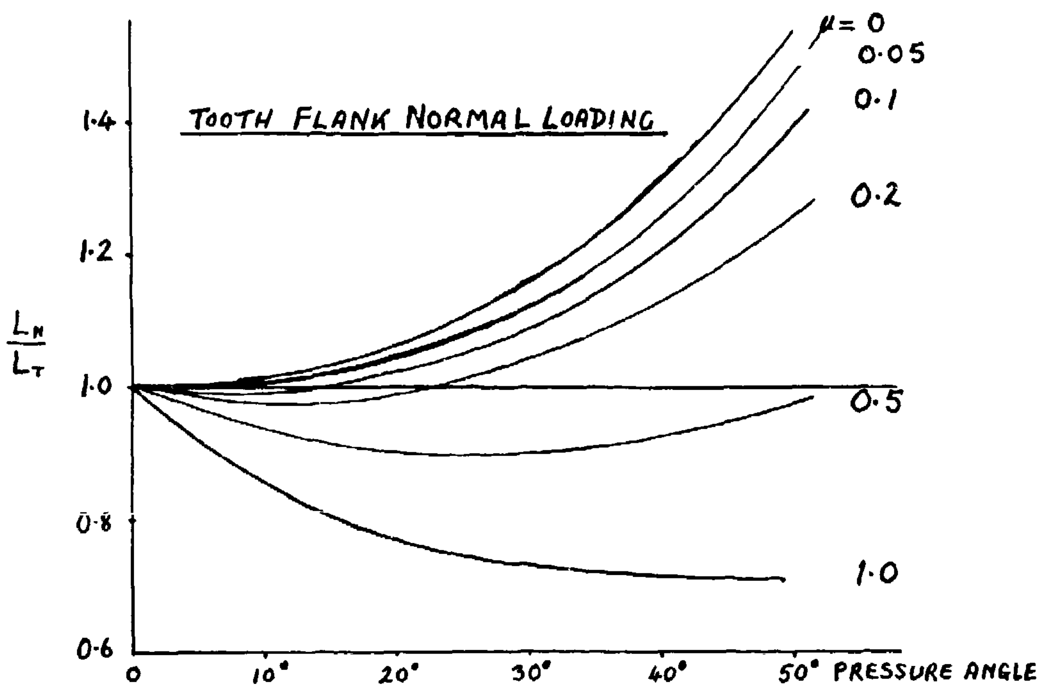


FIGURE A1 a) FRICTIONLESS: b) WITH FRICTION.



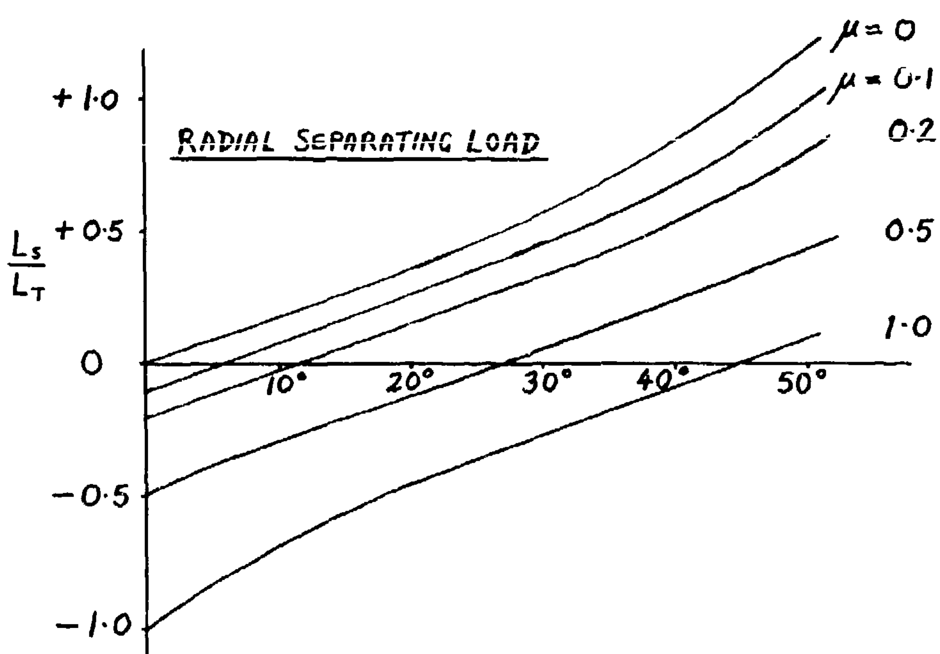


FIGURE A2 THE INFLUENCE OF FRICTION ON TOOTH FLANK LOADS.

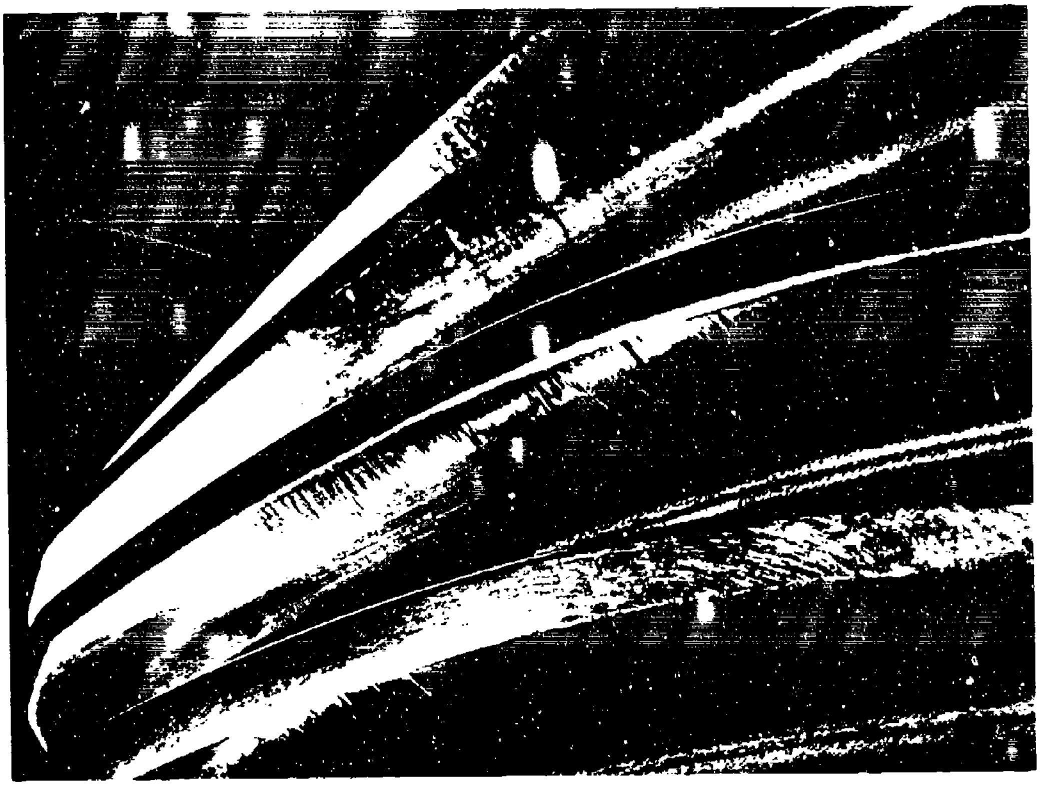


Fig.1 Scuffing damage starting near the tooth tip

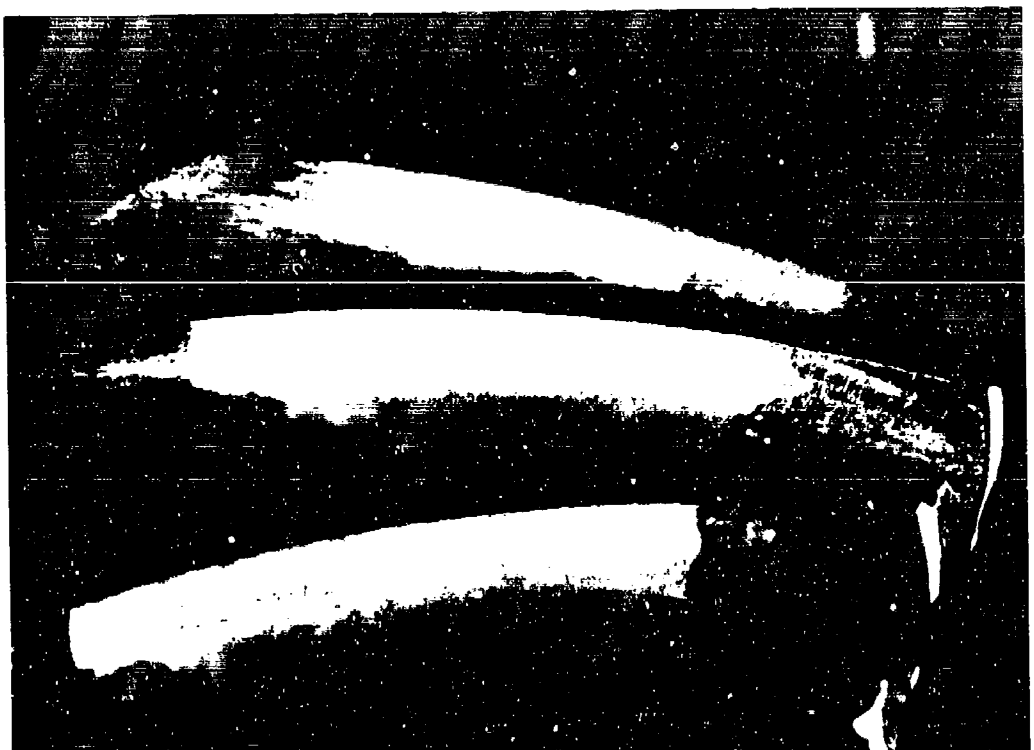


Fig.2 Micro-pitted tooth surface with subsequent scuffing

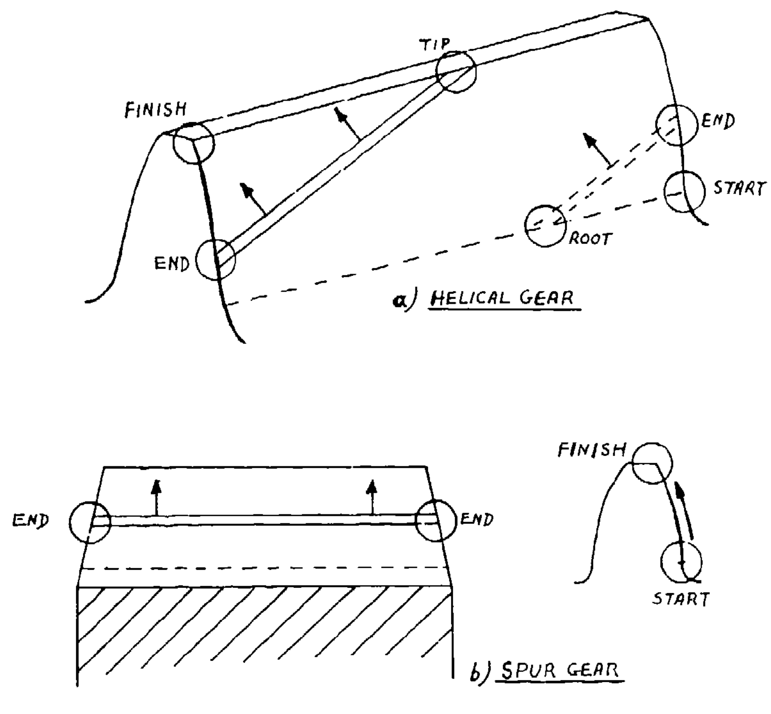


Fig.3 Discontinuities of the line of contact between involute gears

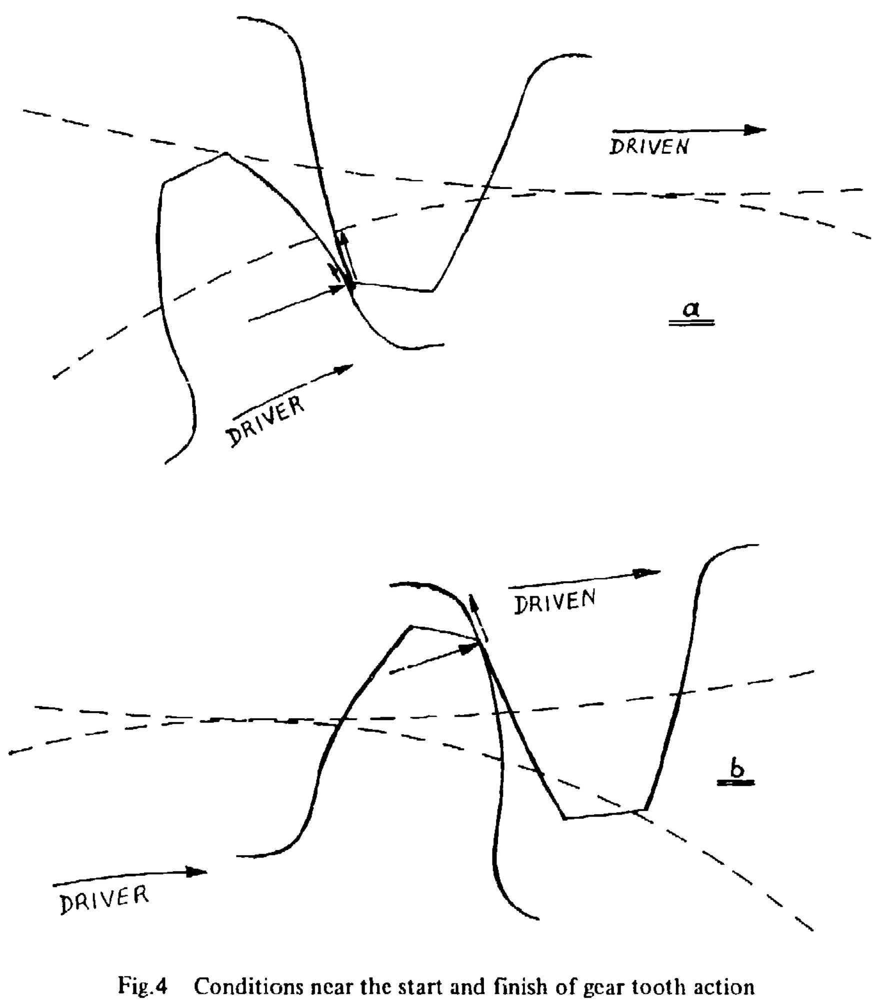


Fig.4 Conditions near the start and finish of gear tooth action

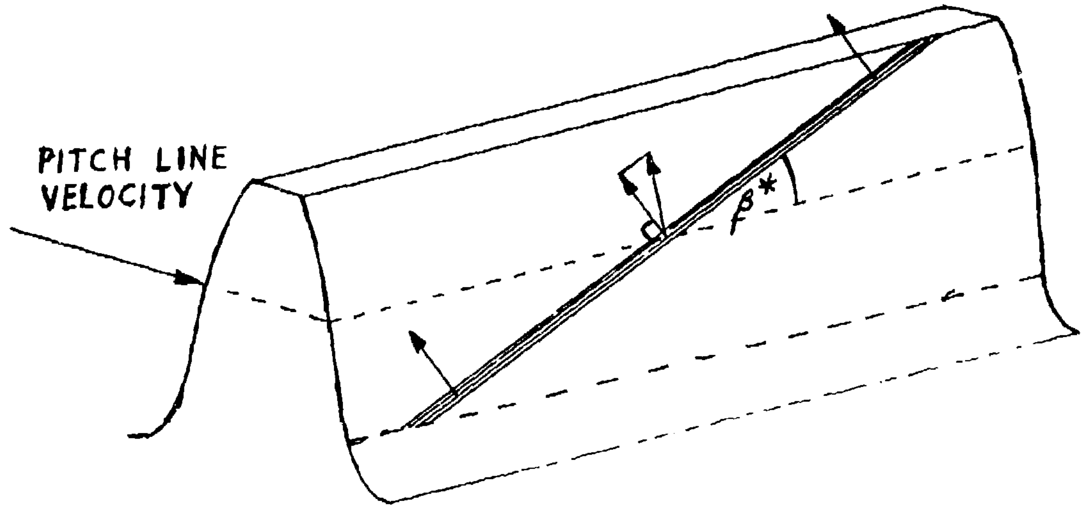


Fig.5 Movement of the contact line over a driving helical involute tooth that generates the oil film

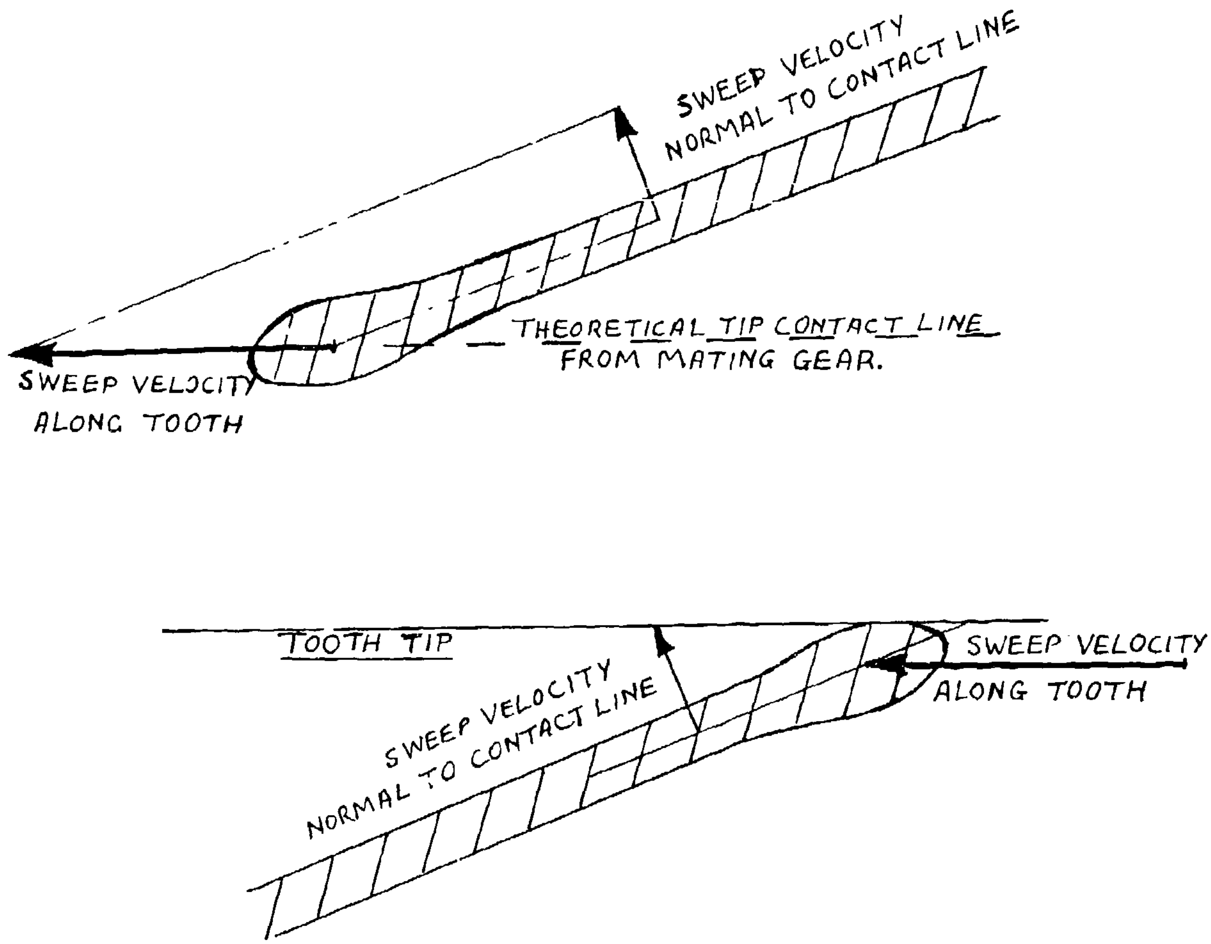


Fig.6 Conditions at the ends of the contact line relating to oil film generation

DISCUSSION

A.Jackson, UK

Could you describe how gearbox life might be improved by giving attention to items discussed in your paper?

Author's Reply

If there is a lubrication breakdown, then the gearing would quickly fatigue and fail. However, with proper lubrication without breakdown, its life could be infinite. Other factors would affect the real life of the gearbox.

ADVANCED TRANSMISSION COMPONENT DEVELOPMENT

by

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SUMMARY

Recently, the emphasis in helicopter gearbox development has concentrated on those design innovations which will permit high temperature operation at increased speeds without degrading strength or weight goals. Several of these design concepts are discussed in this paper. One avenue of investigation, which shows promising strength or weight advantages, is high contact ratio gearing. The means taken to obtain a high contact ratio tends to produce an inherently weaker tooth and reliance must be placed on the multiple load-sharing feature of this design to achieve an advantage over low contact ratio gears. This paper discusses the appropriate consideration which must be addressed in the design stage to achieve optimum results. To provide high temperature capability, two (2) UH-60A helicopter main transmission housings, were fabricated from a stainless steel alloy to replace the conventional magnesium alloy casing. Design details and fabrication procedures are discussed.

INTRODUCTION

Gears. in one form or another, have been in use for about five thousand years and are almost as old as recorded civilization. The discipline of gear design and manufacture has increased in sophistication at an accelerated pace throughout history and has now reached a level of development where it is not unusual for a typical high speed turbine drive pinion to experience over 10 billion contact cycles without breakage or even serious wear after five years of usage.

In the coming decades we can look forward to further advancements in all aspects of gearing development, with increased emphasis placed on such areas as: new gear materials with increased survivability traits, particularly when running at higher temperatures or without oil; and new gear designs with improved dynamic characteristics resulting in reduced noise generation.

High contact ratio gears (HCRG) are gaining wider acceptance in aerospace designs specifically because of beneficial strength and noise attributes. One possible drawback inherent to this configuration, however, is the fact that multiple tooth contacts are necessary to gain an advantage over conventional designs consequently excessive tooth spacing errors must be avoided. Accordingly, special attention should be given not only to the fabrication and processing of HCRG to attain the highest quality gears possible so that the effects of random tooth errors can be minimized, but also to the determination of load sharing among the contacting teeth. This paper presents a systematic approach for obtaining the tooth loads in HCRG.

The analysis presented herein is based on the work conducted by the Sikorsky Aircraft and Hamilton Standard divisions of United Technologies Corporation in support of a NASA-Lewis HCRG evaluation program, NAS3 17859.

Helicopter gearbox housings have traditionally been fabricated from cast magnesium. Unfortunately the strength of this material deteriorates rapidly as operating temperatures increase. Additionally magnesium alloy exhibit relatively poor fatigue strength. This attribute is becoming increasingly important for support structures subject to ground-air-ground (GAG) fatigue loading. The results of a program to evaluate the use of a fabricated stainless steel housing as a replacement for the conventional cast magnesium is presented.

HIGH CONTACT RATIO GEARING

Among the goals of current aerospace research and development programs are increased reliability, and power-to-weight ratio of power transmission systems. Considerable effort has been directed at improving the performance of spur gears with respect to these two criteria. This effort, however, has been primarily aimed at the development of new gear materials; such as VACCO X2, Super Nitralloy, AISI M-50, and CBS 600; and advanced manufacturing methods; such as high-energy-rate forgings, and roll-formed gears. Although some research has been done on alternate gear tooth forms such as the

Wilhaber/Novikov conformal system, Reference (1), the profile geometry of aerospace gearing has remained basically unchanged over the years.

One possibility for improving the performance of spur gears that is gaining wider acceptance to the extent that is is beginning to be used with increased confidence, is high contact ratio gearing (HCRG). Most of the present day spur gearing operate with a contact ratio between 1.2 and 1.6 which means that these designs have a point in the mesh cycle, or along the line of action (Figure 1), where one tooth must take the entire load. HCRG is herein defined as any gear mesh that has at least two tooth pairs in contact at all times, i.e., contact ratio of 2.0 or more. See Figure 2. Because the transmitted load is always shared by at least two pairs of teeth, in this configuration, the individual tooth loading is considerably less for HCRG than for present low contact ratio designs. HCRG, however, inherently requires gear teeth with lower pressure angles, finer diametral pitches, or increased working depths all of which tend to increase the tooth root bending stress per individually applied load. In addition, it would be expected that HCRG would be more sensitive to tooth spacing errors and profile modifications because of the multiple tooth contacts and attendent load sharing requirement.

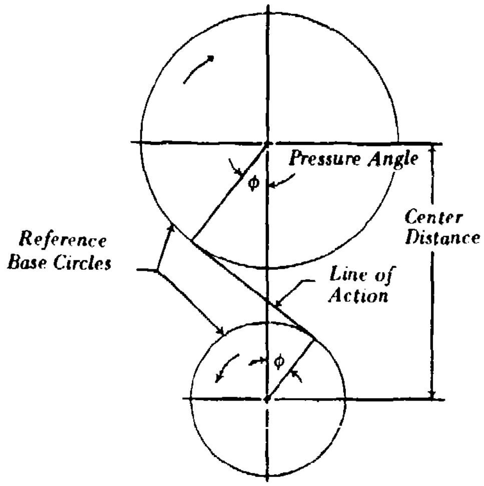

Figure 1 Line of Action for Involute Tooth Pair

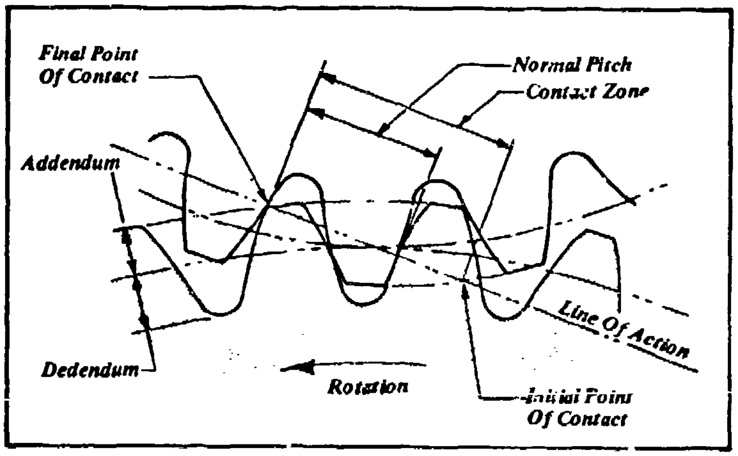

Figure 2a Conventional Tooth Design

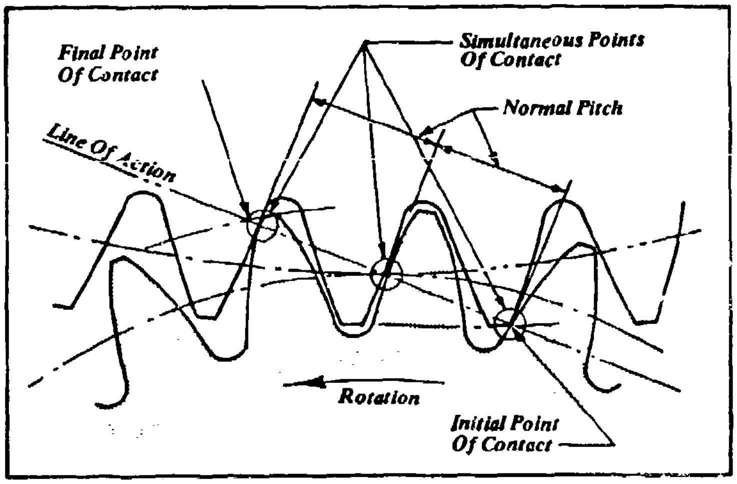

Figure 2b High Contact Ratio Tooth Design

To properly design and evaluate HCRG, three basic analyses are required. First a static analysis is necessary to calculate primary geometric relationships, theoretical contact ratio, relative tooth stiffnesses or compliance values, and the amount of profile modification required to effect a smooth transition of load at the beginning and end of tooth contact. A system dynamic analysis is done next to determine either the total dynamic load, or the dynamic incremental load accruing as a result of random tooth errors and tooth deflections. Finally, a stress sensitivity analysis is performed on the gear teeth to convert the tooth loading into root bending and contact stresses so that these can be compared to acceptable design levels. This last procedure is beyond the scope of this paper but is adequately covered in Reference 5.

Static load analysis

In any design situation there are always certain apriori restrictions imposed on the designer which limits his choice of design variables. Among these may be gear ratio, center distance, maximum pinion size, available tooling, etc. There are five basic design variables which should be considered to achieve a high contact ratio design. The choice, however, may be conveniently narrowed down to two or three because of these prior design restrictions.

The contact ratio for standard involute spur gear teeth may be increased by any of the following means:

1) Lengthening the line of contact by decreasing the angle of the line of action, i.e., lower pressure angles.

2) Lengthening the line of contact by increasing the addendum circle of one or both gears, i.e., increased working depths.

3) Decreasing the normal distance between adjacent teeth, i.e., finer pitch and increased number of teeth.

The contact ratio of a pair of mating spur gears is calculated by:

$$mp = \left(AG\sqrt{Ro_1^2 - Rb_1^2} + AM\sqrt{Ro_2^2 - Rb_2^2} - AG\,AM\,C\sin\phi\right)/Pb \qquad (1)$$

where	mp	=	theoretical profile contact ratio
	Ro_1	=	outside radius of driving gear
	Rb_1	=	base circle radius of driving gear
	Ro_2	=	outside radius of driven gear
	Rb_2	=	base circle radius of driven gear
	C	=	center distance
	ϕ	=	pressure angle
	Pb	=	base pitch
	AG	=	driving gear factor (1 for external gears) (-1 for internal gears)
	AM	=	driven gear factor (1 for external gears) (-1 for internal gears)

This equation can be rewritten more conveniently in the following form:

$$mp = \frac{1}{2\pi\cos\phi}\left[AG\sqrt{(N_1\sin\phi)^2 + 4a_1'(N_1 + a_1')} + AM\sqrt{(mg\,N_1\sin\phi)^2 + 4a_2'(mg\,N_1 + a_2')} - (AG + AM\,mg)\,N_1\sin\phi\right] \qquad (2)$$

where	N_1	=	number of teeth in driving member
	mg	=	gear ratio (mg > 1)
	a_1'	=	one-pitch addendum of driving member
	a_2'	=	one-pitch addendum of driven member

It is obvious from equation (2) that, for design purposes, the five basic factors which can be varied to change the contact ratio are: diametral pitch, number of teeth, gear ratio, gear addenda, and drive-side pressure angle.

While a higher contact ratio can be obtained by increasing gear depth, the longer addendum, if extended too far, can create problems of pointed teeth or undercut tooth flanks. The authors suggest that a topland thickness of .25/Pd is a practical minimum for case-hardened HCRG. A tooth with less topland thickness may be susceptible to edge-chipping and breakage due to through-carburization at the tip. Under ordinary conditions, undercut teeth are also unacceptable because of reduced beam strength. Figures 3 through 6 show contact ratios of external spur gear teeth as a function of pressure angle, addenda modification and number of teeth. The possible design region is limited on the bottom by the undercut limitation and on the right by the minimum topland restriction. These graphs show that the chief limitation on HCRG of low pressure angles is from undercutting. For higher pressure angles, the contact ratio is limited by minimum topland thickness. As can be seen from Figure 6, the minimum

topland requirement effectively eliminates gears with a pressure angle of 25° or more from consideration for high contact ratio gear design. Figure 7 shows that the contact ratio is also virtually insensitive to gear ratio, especially for ratios above 2.0.

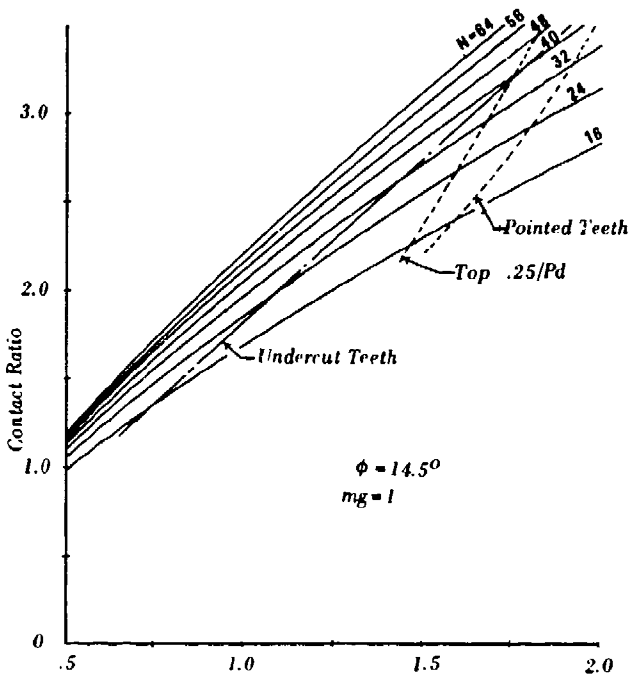

Figure 3 Addenda Modification (aPd)

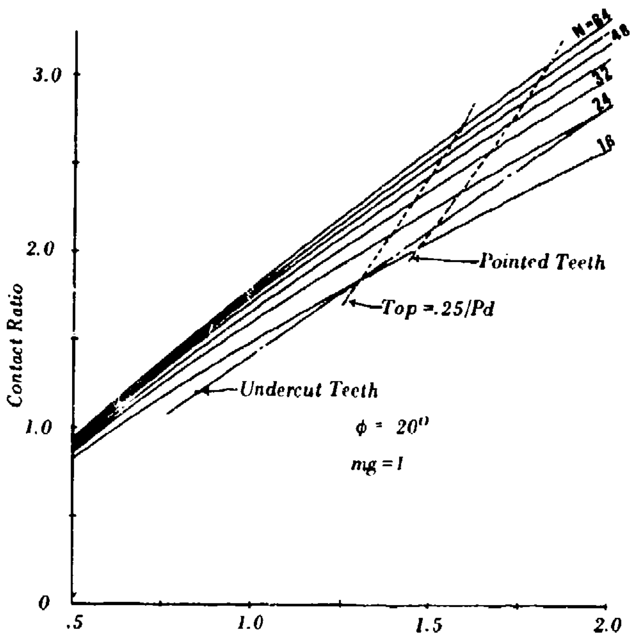

Figure 4 Addenda Modification (aPd)

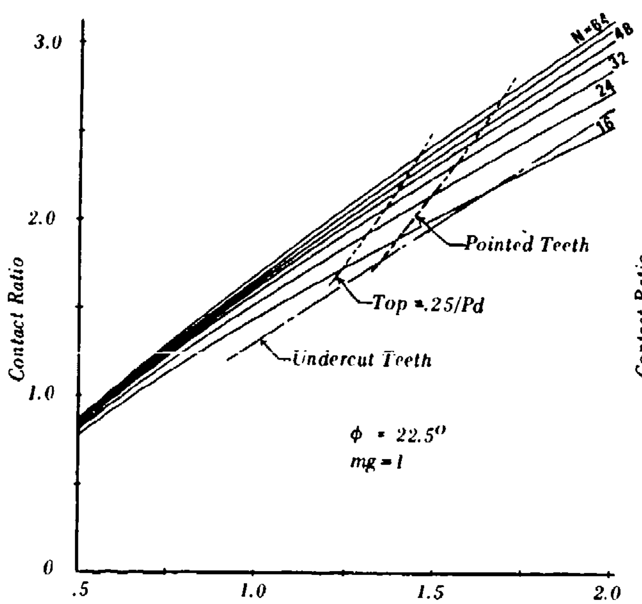

Figure 5 Addenda Modification (aPd)

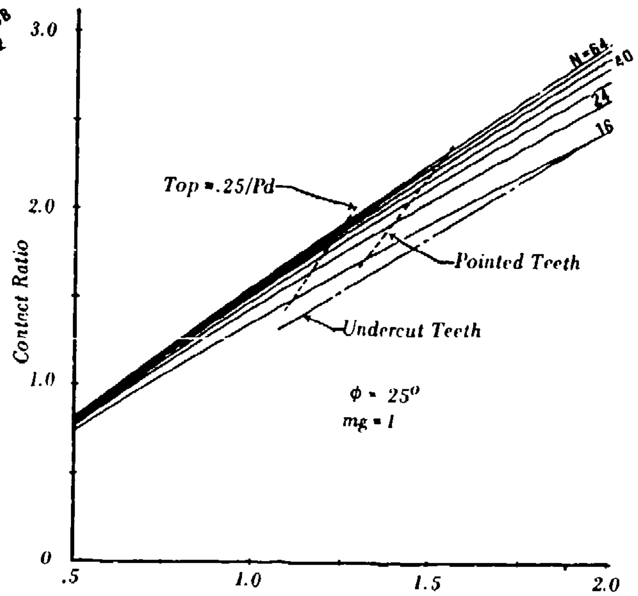

Figure 6 Addenda Modification (aPd)

Tooth pair compliance

The teeth of loaded spur gears deflect a measurable amount which is usually greater than the manufacturing errors in the gear teeth themselves. The effect of this tooth deflection is the same as that of profile or pitch errors. It comprises a major part of the total transmission error and plays an important part in the dynamic action of the loaded gear teeth. It is now well known (Reference 2) that the life of spur gears can be prolonged and the gears made quieter by making a profile modification to the gear tooth to account for this tooth deflection. Calculation of gear tooth deformations on loaded gear teeth, therefore, is an important and necessary step in the gear design process.

The elastic deflection characteristics of an individual gear tooth can best be described in terms of a gear tooth compliance coefficient. This coefficient is defined as the relative deflection, measured normal to the tooth profile and along the line of action, per unit force applied per unit gear face width. Thus:

$$C = \frac{E F \delta}{W} \tag{3}$$

where C is the dimensionless tooth compliance
E is the modulus of elasticity
δ is the normal tooth displacement along the line of action
F is the tooth face width
and W is the normal tooth load

A gear tooth loaded by a concentrated force acting normal to the tooth profile and in the direction of the line of action is subjected to three distinct types of elastic deformations. These are:

1) a displacement of the tooth centerline due to bending and compressive deflection and to transverse shear deflection-(cantileverl beam deflection)

2) a local surface deformation at the point of contact-(Hertzian compression)

3) a rotation and shear deflection at the base of the tooth due to the flexibility of the supporting structure-(foundation effect)

Analytical expressions for determining these displacements are presented below.

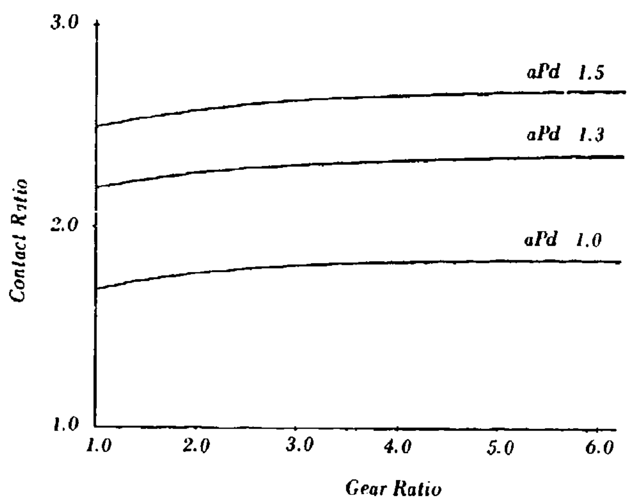

Figure 7 Contact Ratio vs. Gear Ratio

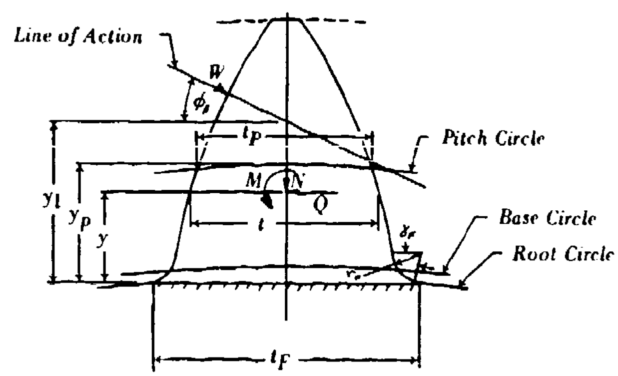

Figure 8 Gear Tooth Bending Compliance Model

Bending deformation

Figure 8 shows the gear tooth model to be analyzed for bending deflection an elastic beam on a rigid foundation. A force W is imposed on the tooth, along the line of action, at a given contact position on the tooth centerline defined by the height y_l above the base. To find the deflection, δ , at the point of load application due to the bending deformations of the tooth, the external work done by the force W is equated to the sum of the internal energies. Thus:

$$\frac{1}{2} W \delta = \frac{1}{2}\int_0^{y_l} \frac{M^2}{EI}\, dy + \frac{1}{2}\int_0^{y_l} \frac{1.2Q^2}{GA}\, dy + \frac{1}{2}\int_0^{y_l} \frac{N^2}{EA}\, dy \tag{4}$$

where referring to Figure 8:

$M = W \cos\phi_N (yl - y)$ - bending moment
$Q = W \cos\phi_N$ - shear load component
$N = W \sin\phi_N$ - radial load component
$I = 1/12\, Ft^3$ -area moment of inertia
$A = Ft$ - cross sectional area
and $G = \frac{E}{2(1+\nu)}$ - shear modules

Substituting these relationships into equation (4) and simplifying, yields the following expression for tooth bending compliance.

$$C_B = 12 \cos^2\phi_N \left\{ \int_0^{y_l} \frac{(y_l - y)^2}{t^3} dy + \left[2(1+\nu) + \frac{\tan^2\phi_N}{12} \right] \int_0^{y_l} \frac{dy}{t} \right\} \tag{5}$$

Note that the tooth thickness t is a complex function of y, and encompasses both the involute profile to its point of tangency with the fillet and a portion of the fillet itself. The integral components of equation (5) cannot be readily evaluated in closed form, however, the equation can be solved quite easily using numerical methods and a digital computer to the desired accuracy.

Hertzian compression

To calculate the Hertzian deformation, the contacting teeth are considered as loaded cylindrical rollers. Referrring to Figure 9; ρ_1 and ρ_2 are the equivalent radii of curature of the profiles at the point of contact. Weber, in Reference 3, developed an expression specifically for the local deformation of two gear teeth. Both teeth are compressed by the force W. It is assumed in this derivation that the compression is from the point of contact in the direction of the applied load up to the centerline of the tooth as indicated by the dimensions h_1 and h_2. This assumption implies that the compression extends to the center of the tooth and then is transmitted as a transverse shear force to the rest of the gear body. Based on this assumption, the Hertzian compression for loaded gear teeth of the same material is given by:

$$\delta_H = \frac{2\,W\,(1-\nu^2)}{\pi F E} \left[\ln \frac{2 h_1}{b} + \ln \frac{2 h_2}{b} - \frac{\nu}{(1-\nu)} \right] \tag{6}$$

where

$$b = \sqrt{\frac{8\,W}{\pi F E} \, \frac{\rho_1 \rho_2}{\rho_1 + \rho_2} \, (1-\nu^2)} \tag{7}$$

the half width of the band of contact.

Note that b is also related to the Hertz stress σ_C by

$$b = \frac{4(1-\nu^2)}{E} \, \frac{\rho_1 \rho_2}{\rho_1 + \rho_2} \sigma_C \tag{8}$$

The tooth pair Hertzian compliance is then

$$C_H = \frac{2(1-\nu^2)}{\pi} \left[\ln \frac{4 h_1 h_2}{b^2} - \frac{\nu}{(1-\nu)} \right] \tag{9}$$

Note that b in equation (9) is a function of the tooth load and that the Hertzian compliance is thus an implicit function of W. The solution of this function can be accomplished by iterative methods coupled with evaluation of the individual tooth loads developed further in this paper. An initial value of Hertzian stress can be assumed and the corresponding value of b calculated from equation (8).

Foundation flexibility

The deflection of the tooth at the load point due to deflection and rotation at its base can be evaluated by treating the tooth as a rigid beam built into an elastic foundation and equating the deforming work done by the load W to the internal stress energies of the support.

Based on the geometry of Figure 8 and the work done by O'Donnell (Reference 4) in the evaluation of influence coefficients; the foundation compliance can be shown to be:

$$C_F = \cos^2\phi_N (1-\nu^2) \left[\frac{16.67}{\pi} \left(\frac{y_l}{t_F}\right)^2 + \frac{2(1-2\nu)}{(1-\nu)} \frac{y_l}{t_F} + \frac{4.82}{\pi} \left(1 + \frac{\tan^2\phi_N}{2.4\,(1+\nu)} \right) \right] \tag{10}$$

It can be seen from this equation that the foundation flexibility is primarily a function of load position and the tooth thickness at the built-in section. The location of the base and the thickness t_F is conveniently defined by a fillet angle γ_F. O'Donnell recommends a constant value of approximately 75° for γ_F. At this point a question arises concerning the magnitude of γ_F and the location of t_F. Cornell, Reference (5), takes a more rational approach and calculates the value of γ_F which maximizes the total deforming work (bending + foundation) done on the tooth. Figure 10, reproduced from Reference 5, shows the variation of fillet angle with load position and fillet radius to obtain maximum beam flexibility. It can be assumed, then, that the fillet angle should not be considered a constant, but varying with load position. Using the above procedures, and an iterative routine to locate the position of maximum

flexibility, the support compliance can be calculated for any position along the line of action.

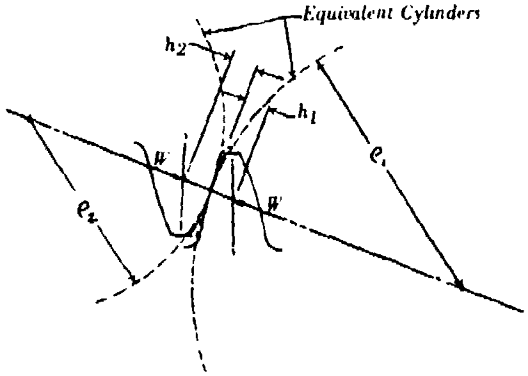

Figure 9 Gear Tooth Hertz Compliance Model

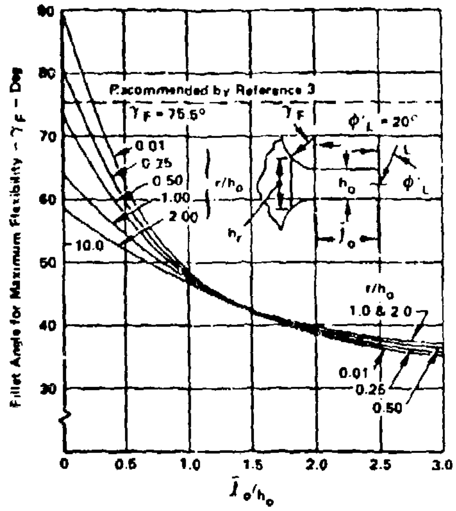

Figure 10 Fillet Angle for Maximum Flexibility vs Load Point from Fillet

Total tooth pair compliance

The total tooth pair compliance can now be found as a function of load position from the initial point of contact to the final point of contact. The total compliance is the sum of the individual compliances.

$$C_T = C_{B1} + C_{B2} + C_H + C_{F1} + C_{F2} \tag{11}$$

Thus the total compliance that exists at any phase of tooth engagement can be formulated readily for a single pair of meshing teeth. The compliance curve will be concave upward showing a characteristic stiffening of the tooth pair near the pitch point. For a gear ratio of unity, the compliance curve will be symmetrical about the pitch point. For other ratios, the compliance at the tip of the larger gear of the tooth pair will be smaller than the compliance at the tip of the smaller gear.

Static load sharing in multiple contact teeth

In HCRG, the transmitted load is shared by two or more teeth. To properly design these gears, the proportionate load sharing between the loaded tooth pairs must be determined with some degree of accuracy.

A typical compliance curve for a HCRG design with a contact ratio of 2.32 would be shown in Figure 11. The non-dimensional compliance is plotted as a function of the normalized position of the contact point along the line of action. Referring to Figure 11, the tooth pair engage at a and disengage at f. Since the actual distance between these two points is the length of the path of contact, the normalized distance af is numerically equal to the contact ratio.

Assuming that at a particular point in the mesh cycle, there are three tooth pairs in contact; and

$W_1, W_2, W_3,$ = the normal loads on the successive tooth pairs

$C_1, C_2, C_3,$ = the mesh compliance of the tooth pairs at the positions of simultaneous contact

$e_1, e_2, e_3,$ = unloaded relative position error of the mating teeth at points of contact. (a positive value indicates a gap and a negative value denotes an interference)

$\epsilon_1, \epsilon_2, \epsilon_3,$ = amount of profile modification at the specific points of contact (a positive value or zero)

Note that subscripts 1, 2, and 3 refer to the first, second and third tooth pairs in contact, each separated by one base pitch.

When multiple tooth pairs are in contact, the total transmission error at each contact point must be equal to avoid interference. Equating this transmission error of each contacting tooth pair, we have for three pair contact:

$$\frac{W_1C_1}{E\,F} + e_1+\epsilon_1 = \frac{W_2C_2}{E\,F} + e_2+\epsilon_2 = \frac{W_3C_3}{E\,F} + e_3+\epsilon_3 \tag{12}$$

also note that

$W_1 + W_2 + W_3 = W_N$, the total normal transmitted load.

Solving these equations for the individual tooth loads we get

$$W_1 = \frac{W_NC_2C_3 - E\,F\,[(e_1+\epsilon_1)(C_2+C_3) - (e_2+\epsilon_2)C_3 - (e_3+\epsilon_3)C_2]}{C_1C_2+C_1C_3+C_2C_3} \tag{13}$$

$$W_2 = \frac{W_NC_1C_3 - E\,F\,[(e_2+\epsilon_2)(C_1+C_3) - (e_3+\epsilon_3)C_1 - (e_1+\epsilon_1)C_3]}{C_1C_2+C_1C_3+C_2C_3} \tag{14}$$

$$W_3 = \frac{W_NC_1C_2 - E\,F\,[(e_3+\epsilon_3)(C_1+C_2) - (e_1+\epsilon_1)C_2 - (e_2+\epsilon_2)C_1]}{C_1C_2+C_1C_3+C_2C_3} \tag{15}$$

Similar equations can be derived for the zones of two-pair contact. The results are:

$$W_1 = \frac{W_NC_2 - E\,F\,[(e_1+\epsilon_1) - (e_2+\epsilon_2)]}{C_1+C_2} \tag{16}$$

$$W_2 = \frac{W_NC_1 - E\,F\,[(e_2+\epsilon_2) - (e_1+\epsilon_1)]}{C_1+C_2} \tag{17}$$

These equations allow one to calculate the individual static tooth loads at any mesh position when the compliance, error, and profile relief relationships are known.

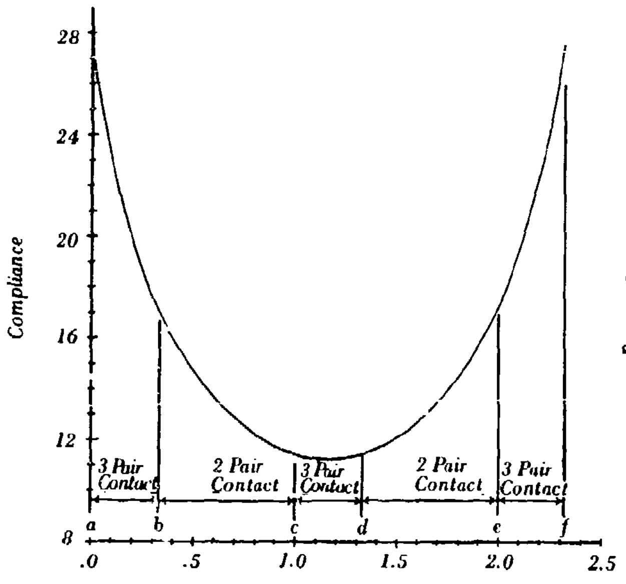

Figure 11 Tooth Pair Compliance

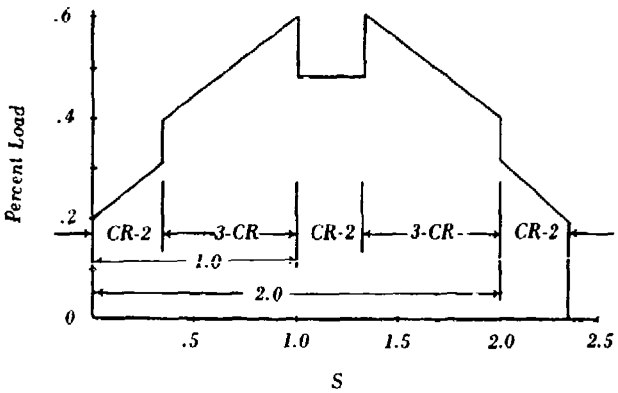

Figure 12 Load Distribution without Tip Relief

Assuming, for example, that the teeth are unmodified and the tooth profiles are perfect involutes, the tooth load distribution on an individual tooth can be calculated as it proceeds through mesh. For the case cited;

$e_1 = e_2 \quad e_3 = 0$

$\epsilon_1 = \epsilon_2 = \epsilon_3 = 0$

For the 3 pair contact zone;

$$W_1/W_N = \frac{C_2C_3}{C_1C_3 + C_1C_2 + C_2C_3} \tag{18}$$

$$W_2/W_N = \frac{C_1C_3}{C_1C_3 + C_1C_2 + C_2C_3} \tag{19}$$

$$W_3/W_N = \frac{C_1C_2}{C_1C_3 + C_1C_2 + C_2C_3} \tag{20}$$

and for the two-pair contact zone;

$$W_1/W_N = \frac{C_2}{C_1 + C_2} \tag{21}$$

$$W_2/W_N = \frac{C_1}{C_1 + C_2} \tag{22}$$

Using these relationships, the tooth load distribution on an individual tooth can be plotted showing the variation in load as the contact moves along the line of action. Using the compliance values of Figure 11, such a load diagram is shown in Figure 12. Note from this diagram that at the beginning and end of contact, and whenever the load shifts from 3 pair contact to 2 pair contact, and vice versa; there is a discontinuity in the load distribution. Associated with these abrupt changes in tooth load, there will be corresponding changes in transmitted angular velocity which would adversely affect gear performance.

The purpose of gear tooth profile modification now becomes quite evident. Profile modification is a means of intentionally altering the involute profile to give a closer approach to uniform angular velocity transmission by eliminating the sudden abrupt changes in tooth load. The first step in this process is to reduce the load at the initial and final points of contact to zero. Consider the tooth when it first comes into contact at a. At this time, simultaneous contacts are also occurring at c and e. Assuming again that the gears are error free and that no modification is to be applied below c on either gear, equation 13 becomes;

$$W_a = \frac{W_N C_c C_e - E\,F\,[\,\epsilon_a\,(C_c + C_e) - \epsilon_e C_c\,]}{C_a C_e + C_a C_c + C_c C_e} \tag{23}$$

equating this to zero and solving for ϵ_a:

$$\epsilon_a = \frac{W_N/E\,F\,C_c C_e + \epsilon_e C_c}{C_c + C_e} \tag{24}$$

similarly,

$$\epsilon_f = \frac{W_N/E\,F\,C_b C_d + \epsilon_b C_d}{C_b + C_d} \tag{25}$$

Let the form of the profile modification be represented by:

$$\epsilon = \epsilon_a\left(1 - \frac{S}{l_a}\right)^{\alpha} \tag{26}$$

and

$$\epsilon = \epsilon_f\left(1 - \frac{CR - S}{l_f}\right)^{\alpha} \tag{27}$$

where S is the normalized distance from the initial point of contact and l is the length of the modification as shown in Figure 13..

then

$$\epsilon_a = \frac{\frac{W_N}{EF}\;C_c C_e + \epsilon_f\left(1 - \frac{CR-2}{l_f}\right)^{\alpha} C_c}{C_c + C_e} \tag{28}$$

$$\epsilon_f = \frac{\frac{W_N}{EF} C_b C_d + \epsilon_a \left(1 - \frac{CR-2}{l_a}\right)^\alpha C_d}{C_b + C_d} \tag{29}$$

note that the bracketed terms in equations (28) and (29) mus be either positive or zero.

Solving these two equations for ϵ_a and ϵ_f

$$\epsilon_a = \frac{W_N}{EF} \times \frac{C_c C_a (C_b + C_d) + C_b C_c C_d \left(1 - \frac{CR-2}{l_f}\right)^\alpha}{(C_c + C_a)(C_b + C_d) - \left(1 - \frac{CR-2}{l_a}\right)^\alpha \left(1 - \frac{CR-2}{l_f}\right)^\alpha C_c C_d} \tag{30}$$

$$\epsilon_f = \frac{W_N}{EF} \times \frac{C_b C_d (C_c + C_a) + C_c C_d C_a \left(1 - \frac{CR-2}{l_a}\right)^\alpha}{(C_c + C_a)(C_b + C_d) - \left(1 - \frac{CR-2}{l_a}\right)^\alpha \left(1 - \frac{CR-2}{l_f}\right)^\alpha C_c C_d} \tag{31}$$

If the start of the tip relief is limited to the tip of the tooth, outboard of b, the above equations simplify to:

$$\epsilon_a = \frac{W_N}{EF} \frac{C_c C_a}{C_c + C_a} \tag{32}$$

and

$$\epsilon_f = \frac{W_N}{EF} \frac{C_b C_d}{C_b + C_d} \tag{33}$$

and the profile relief, as a function of S is given by:

$$\epsilon = \frac{W_N}{EF} \frac{C_c C_a \left(1 - \frac{CR-S}{l_a}\right)^\alpha}{C_c + C_a} \qquad 0 \leq S \leq l \tag{34}$$

$$\epsilon = 0 \qquad l_a \leq S \leq (CR - l_f) \tag{35}$$

$$\epsilon = \frac{W_N}{EF} \frac{C_b C_d \left(1 - \frac{S}{l_f}\right)^\alpha}{C_b + C_d} \qquad (CR - l_f) \leq S \leq CR \tag{36}$$

Figure 13 Profile Relief Distribution

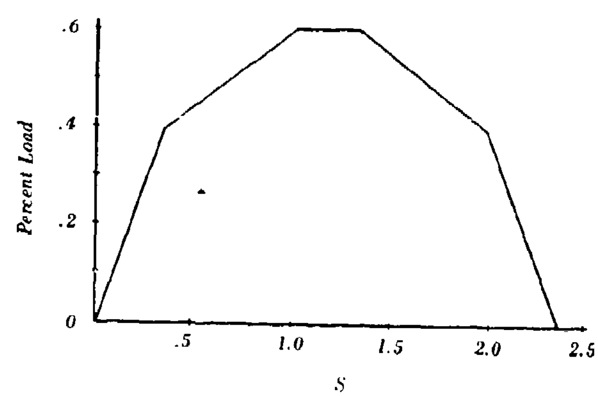

Figure 14 Load Distribution with Tip Relief

When these profile deviations are substituted into equations 13, 14, 15, 16 and 17; the individual tooth loads for any value of S can be determined. The resulting change in the load diagram will be as shown in Figure 14. Thus the profile modification effectively removes the load discontinuities and results in a smoother load transition and a more uniform velocity transmission. The actual shape of the load distribution curve in the three-pair regions depends upon the value of the exponent and the modification length l. Figure 15 shows the effect of varying these two parameters. It is apparent from this figure that l has more influence on the load distribution than does α.

The effect of tooth errors on the tooth load can be handled in much the same fashion but it is inadvisable to try to correct for index errors by profile relief since they are random in nature and unpredictable with respect to sign. When a tooth is out of

position with respect to adjacent teeth, it will take more or less load depending upon whether the error is positive or negative. The resulting load distribution for an error of EFe/Wn = -.85 is shown in Figure 16.
It can be seen, from this diagram, that the maximum load increases by about 6% due to a tooth spacing error which is equivalent to approximately .0002 in.

Dynamic load analysis

The effect of profile modification and random tooth errors on dynamic tooth loads can be estimated most simply by applying an appropriate dynamic multiplying factor to the static loads. Since the dynamic load really is an incremental load superimposed on the transmitted load, this approach is subject to error, particularly at the low load portion of the spectrum.
A more systematic approach would be to calculate the total dynamic load using a suitable dynamic tooth model which takes into account both tooth profile modification and tooth errors. Such a model, used in the analysis of Reference 5, is shown in Figure 17. This model assumes that the two mating gears act as the variable springs of a dynamic system excited by the non-linear meshing action and stiffnesses of the gear teeth.

Referring to Figure 17, the differential equation of motion between the two gears can be written as

$$M\ddot{S}_r + d\dot{S}_r + \sum_{j=1}^{n} W_j = W_N \qquad (37)$$

where Wj is the individual tooth load, W_N is the total applied load, $S_r = S_2 - S_1$ the relative motion between the two gears, d is the damping factor, and n is the number of tooth pairs in contact. This equation in dimensionless form is:

$$\ddot{y} + 2\xi\dot{y} + \sum_{j=1}^{n} W_j/W_N = 1 \qquad (38)$$

where

$y = S_r k_0/W_N$ $\quad$ k_0 = stiffness at pitch point

$\dot{y} = dy/dt$ $\quad$ $\omega = \sqrt{k_0/M}$

and ξ is the damping coefficient.

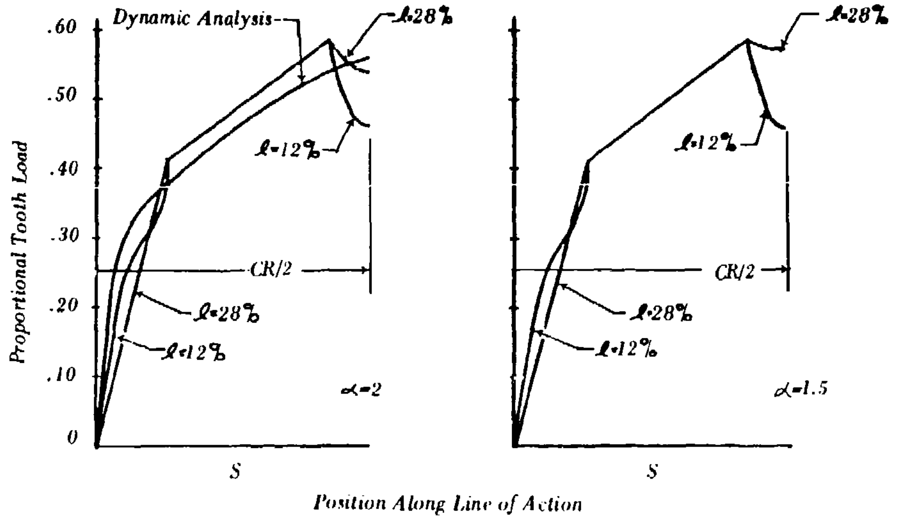

Figure 15 Effect of α and ℓ on Tooth Load

Because of the variable tooth pair stiffnesses acting during the mesh, differential equation (38) is non-linear so that the closed form solutions will apply for only an instant in time and are thus piecewise continuous through the mesh. To solve this equation a time history solution is used, stepping through the mesh cycle in small dimensionless time increments during which the tooth stiffness is assumed to be constant. In this way solutions can be obtained for the deflections and resulting dynamic tooth loads. Implicit in this procedure however, is the knowledge of the initial

values of $\dot{y}$ and y. These can be obtained by interaction on the basis that after the passage of one gear mesh ($\Delta S = 1$), the gear displacement and velocity must be the same as that for the starting condition, when no gear errors are involved. The solution for the case with tooth errors is done then in two steps; first, the iterative solution is obtained for the zero-error case as discussed above. Then starting with the known initial conditions of $\dot{y}$ and y, the analysis is run on a consecutive mesh basis, introducing the specific error on any tooth or teeth desired. A more complete discussion of the details of this procedure is contained in Reference (5).

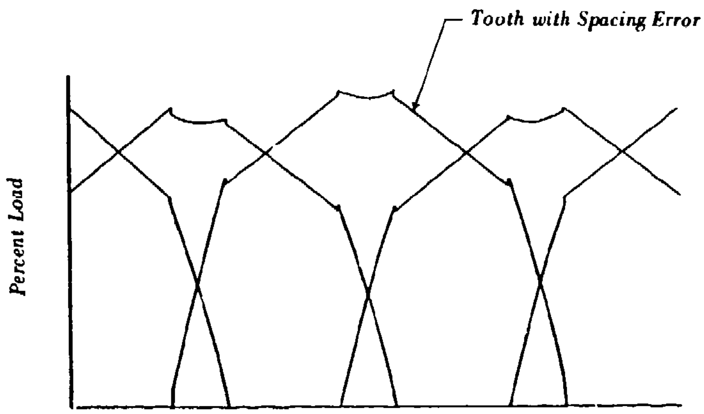

Figure 16 Effect of Tooth Error

The results of a dynamic analysis accomplished by the procedure described above, using the same tooth design and spacing errors is shown in Figure 15. These results show that, for the particular speed used, the dynamic analysis predicts lower loads than the static analysis in the two-pair region and higher loads in the three-pair region. Different results might be expected, however, at other speeds, at other values at profile relief, or at lower damping.

Thus the load distribution in a HCRG design can be adequately calculated using either a static or dynamic analysis for a given profile modification and error condition. A NASA-Lewis sponsored test program is now in progress at Sikorsky Aircraft to experimentally evaluate the above design procedure. Results will be available early in 1982.

Stress sensitivity

Many methods for calculating the stress sensitivity of spur gear teeth are available. See References (6), (7), and (8). The applicability of these methods to HCRG was evaluated in Reference (5). It was concluded that a modified Heywood analysis produced results which compared most faborably with test results.

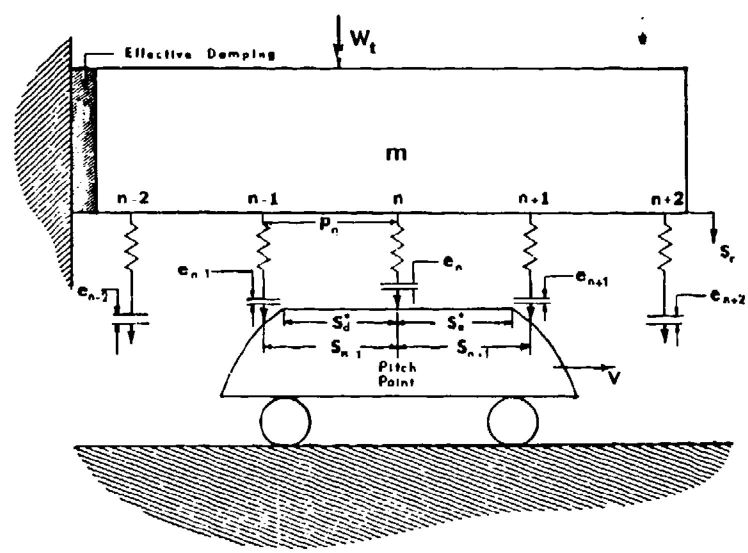

Figure 17 High Contact Ratio Dynamic Load Model

Buttress tooth form

As has been discussed previously, the HCR tooth configuration is inherently weaker than a conventional low contact ratio for the same load design, and any strength advantage obtained is the direct result of multiple load sharing between successively contacting tooth pairs coupled with adequate profile relief and error consideration. One means, presently being tested at Sikorsky, for regaining some of this lost strength while retaining the high contact ratio feature, is the use of a buttress tooth form. Stated simply, buttressed teeth have a larger pressure angle on the coast side of the teeth than on the drive side resulting in the asymmetric tooth form shown in Figure 18. This design has a thicker tooth section at the base of the tooth where the bending stresses are concentrated, albiet with a smaller fillet radius and reduced topland thickness. The net result of this buttressing reflect, however, is a stronger tooth for the same contact ratio. The percent improvement in strength factor is shown in Figure 19 for a drive-side pressure angle of 20° and various coast-side pressure angles, and for three load-application points.

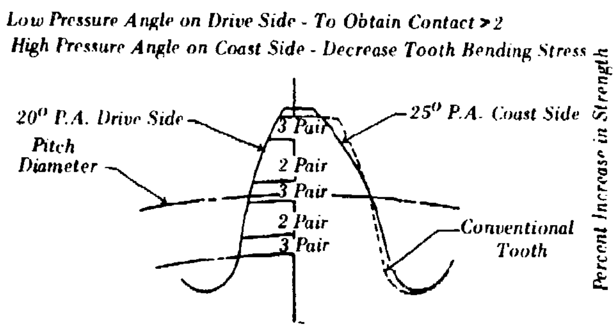

Figure 18 Buttress Tooth Form

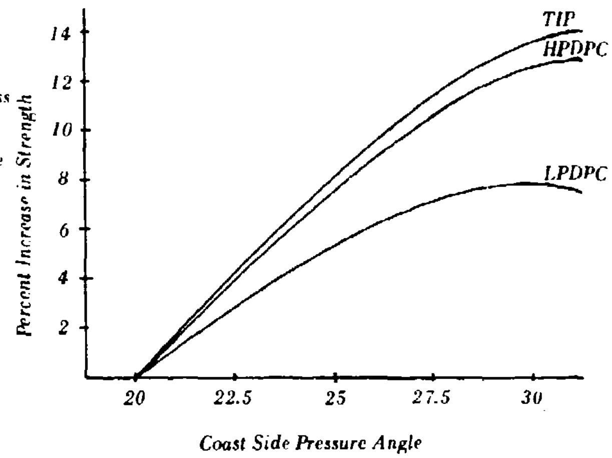

Figure 19 Strength Increase of Buttress Teeth

Constant relative radius of curvature

One of the disadvantages claimed of the involute gear tooth form is the rapidly diminishing radius of curvature of the tooth profile in the vicinity of the base circle. The relative radius of curvature at any contacting point is given by:

$$R_e = \frac{\rho_1 \rho_2}{\rho_1 + \rho_2} \tag{39}$$

where ρ_1 and ρ_2 are the radii of curvature of the respective teeth at the contact point. It can be seen, from this relationship, that the relative radius of curvature is always smaller than that of the smaller gear. Thus as the point of contact approaches the base circle (as it must for HCRG), the relative radius of curvature approaches zero. Since the contact stress between the contacting gear teeth is inversely proportional to the relative radius of curvature, it becomes increasingly large as the radius of curvature becomes smaller. Not only does the contact stress increase as contact approaches the base circle, the sliding velocity increases also, and the scoring hazard, which is a function of the product of contact stress times sliding velocity, can be proportionately larger for HCRG than for conventional gears.

To correct these disadvantages, it is possible to design and fabricate teeth that deviate from an involute curve to the extent that the individual tooth curvatures will give essentially a constant relative radius of curvature across the entire working profile of the tooth. This can be accomplished by using an S-shaped line of action which can be generated by a similarly-shaped hob. This approach will reduce the contact stress at the extreme points of contact thus reducing the scoring hazard without affecting the contact ratio.

A further advantage of the constant relative radius of curvature (CRC) concept is shown in Figure 20. The interferance point is defined by the intersection of a radial line passing through the gear center and normal to the line of action. In the CRC gears, this point is closer to the gear center and thus, the practical minimum number of teeth on the smaller gear can be reduced and the undercuting limitation virtually eliminated. Thus the CRC tooth form has two advantages -- 1) lower Hertz stress will occur at the tips of the teeth where scoring hazard is highest and 2) fewer numbers of teeth are possible on the driving member resulting in higher possible single-mesh ratios.

The effect of the CRC tooth form on load sharing and tooth bending can be addressed by calculating the deviations of the CRC profile from the involute profile and introducing them as profile deviations in equation (13), (14), (15), (16), and 17. This potential modification to the HCRG design will be evaulated at Sikorsky in the near future.

FABRICATED TRANSMISSION HOUSING

Sikorsky Aircraft has been pursuing the fabricated housing for high temperature operation since the early 1970's. An ATL funded study showed that an appreciable weight savings could be achieved through fabrication of a steel or titanium main transmission housing. An in-house effort then resulted in the fabrication of a tail gearbox housing for the RH-53 helicopter. This stainless steel housing, Figure 21, resulted in a 10% weight savings for the same stiffness as the production magnesium housing.

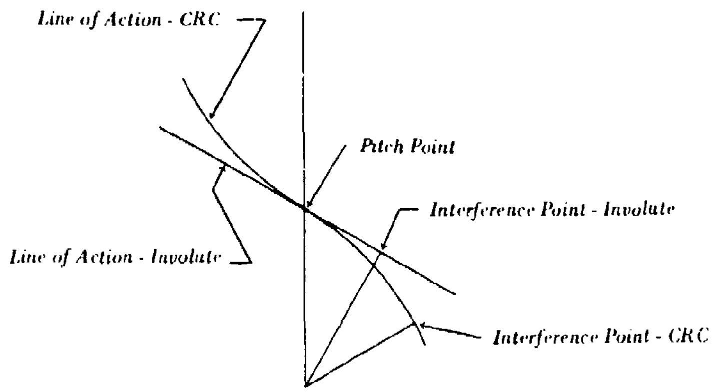

Figure 20 Comparative Interference Points - CRC vs. Involute Profiles

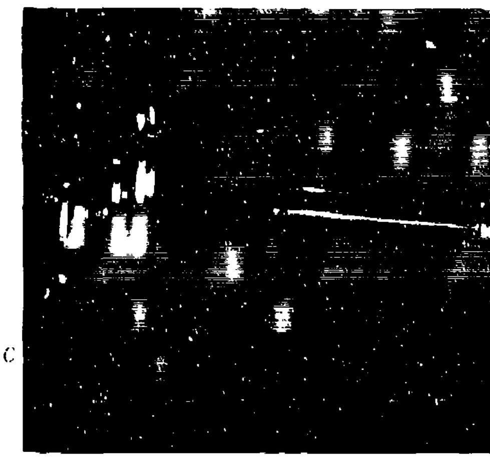

Figure 21 RH-53 Fabricated Tail Gearbox Housing

Material selection

Stainless steel was chosen in these studies for its strength, ease of fabrication, and welding characteristics. Other materials were evaluated, however none measured up to the overall attraction of stainless steel for continuous operation in the 450° - 550°F range.

Aluminum alloys 6061-T6 and 7175-T736 are not viable candidates for advanced transmissions because the operating temperature of the gearbox without an oil cooler is above the aging temperature of aluminum. Magnesium alloys, although 100° higher in aging temperature than aluminum, are still too close to the operating temperature for long term stability. Two high-temperature magnesium alloys do exist but corrosion resistance is low.

Titanium alloys are viable candidate housing materials, however, the higher cost of the end-product does not appear worth the small weight savings when compared to stainless steel.

Composites offer the promise of good ballistic damage resistance, and resins exist which will resist temperature and synthetic oils (polyimids), but the low thermal conductivity across the resin (in the order of 0.3 BTU/ft/hr ft^2°F) does not allow sufficient heat dissipation by radiation to eliminate the oil cooler. Also, this material does not appear to have the weight savings potential of a stainless steel gearbox without an oil cooler.

Carpenter Custom 450 was selected from stainless steels such as 15-5, 17-4, and 17-7. In the final analysis, two parameters determined the selection of the alloy. First, housing stiffness requirements are more stringent than strength thus allowing the use of a lower-strength material with increased fracture toughness. Second, Carpenter Custom 450 appears more amenable to welding processing than any other alloy. Carpenter Custom 450, a minor modification of 17-4 PH, has the required strength and stiffness properties and improved welding characteristics needed for high volume production.

The chemical compositon of Carpenter Custom 450, given in Table 1, has been balanced to achieve a martensitic structure in all conditions of heat treatment. The carbon content of the alloy is extremely low. This results in a tough, ductile martensite which resists cracking in the weld metal and the heat-affected zone of the parent metal and, because of its low carbon content, preheating prior to welding is not necessary.

Housing design

The design goal for this program was a housing stiffness equal to or better than the BLACK HAWK magnesium housing, Figure 22, and a 10% weight advantage. Since the specific stiffness E/ρ, is basically the same for titanium, steel, aluminum and magnesium, the weight saving was to be achieved by more efficient design, i.e., without the limitations imposed by current casting practices.

Carbon	*0.05 max*	*Chromium*	*14.00/16.00*
Manganese	*1.00 max*	*Nickel*	*5.00/7.00*
Silicon	*1.00 max*	*Molybdenum*	*0.50/1.00*
Phosphorus	*0.03 max*	*Copper*	*1.25/1.75*
Sulfur	*0.03 max*	*Columbium*	*8 x C min*

Table 1 Carpenter Custom 450 Composition-Percent

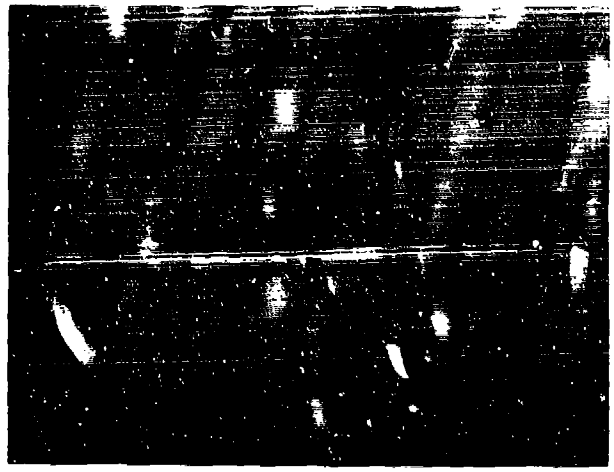

Figure 22 BLACKHAWK Cast Magnesium Housing

The fabricated housing, Figure 23, is of semi-monocoque design and is comprised of a sheetmetal shell to which stiffening ribs are welded. The ribs provide the supports for the inputs and tail take-off and transfer the main rotor bearing loads into the airframe. The shell reacts the shear loads and retains the lubricating oil.

Finite element analysis (NASTRAN) was used to design the housing to withstand a simulated forward crash load of 20 g's without undergoing catastrophic failure. Constraints imposed by having to retrofit existing hardware prevented optimizing the shell to carry this 20g load (equivalent to 67,400 lbs at the rotor centroid.) The addition of ribs to the shell provided the primary path for the main rotor loads and for attachment of the housing to the airfrme. For a forward crash condition, the fore and aft ribs are the primary members. Maintaining structural integrity of these members prevents catastrophic type failure. Whereas magnesium housings, if tested to destruction, would fail in a brittle static fracture mode, a steel housing will fail by buckling and/or instability.

The NASTRAN model used for the stability analysis is shown in Figure 24. This model contains 8963 degrees of freedom and is made up of primarily the CQUAD 4 isoparametric membrane bending quadilaterial elements. Plane stress elements were used for the braces and tail rotor take-off.

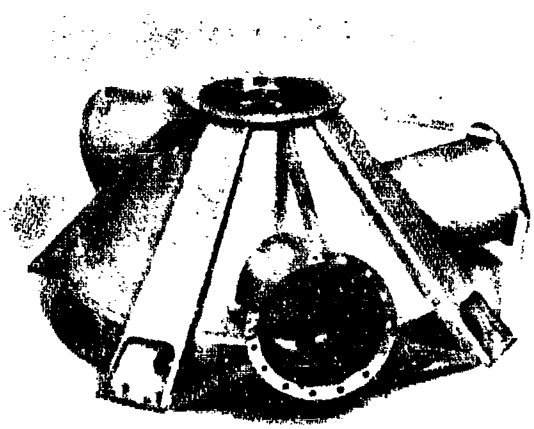

Figure 23 Fabricated Steel Housing

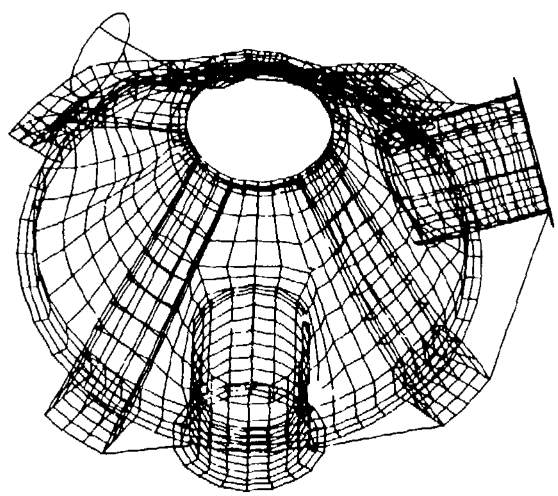

Figure 24 NASTRAN Model-Fabricated Housing

With the load applied at the centroid of the rotor system, the stability analysis revealed three distinct buckling modes occurring in the skin up to the 20g load. As NASTRAN is a linear analysis, post buckling analysis is not possible, however the assumption is made that load redistribution due to buckling is not severe since it is assumed the ribs are carrying the majority of the load. Based on this assumption, the stresses in the housing were evaluated as shown in the ISO-stress contours of Figure 25.

Two housings were fabricated at Sikorsky Aircraft. A special welding fixture was designed to permit logical assembly of detail parts for welding. The bearing journals and flange details were machined from ring forgings and .060 inch sheet stock was butt-welded to form the cylinders and cone. Assembly was by manual gas tungsten arc welding using strips of base material as filler. The heat treatment of the housing was performed on a holding fixture and consisted of a vacuum solution anneal at 1900°F for 15 minutes followed by an aging cycle of 1025°F for 4 hours.

A complete dimensional inspection was performed after welding and aging to check material dimensional changes and distortion. It was observed that subsequent to the aging cycle, a dimensional shrinkage occurred, however, an absolute value of a shrinkage coefficient could not be established. The amount of shrinkage is dependent not only on the material characteristics but also on the structural configuration (restraint offered by the massive rings on the thin sheet metal).

Final machining was accomplished, after heat-treatment, utilizing production tooling and machinery. To prevent excessive chatter and possible pick-up and seizure of the material, Rigidetex, a plastic compound, was molded to the input housings. The dampening characteristics of this material eased machinability of the input bores to the close alignment tolerances required for successful operation of the input bevel pinions and gears. The housing was vapor blasted and passivated to achieve the desired matt finish for optimum heat-rejection and corrosion resistance.

The fabricated housing eliminated the requirement for hardened liners thereby eliminating secondary machining operations. Through-bolts are used exclusively, thereby eliminating the requirement for added material and machining for threaded steel inserts. This resulted in attainment of a 8% savings for the fabricated housing which is estimated to weight approximately 102 lbs in its final configuration compared to the cast magnesium housing weight of 110 lbs.

Thus, the fabricated stainless steel housing offers a viable alternative to the a magnesium alloy casting particularly where high temperature capability and survivability are design requirements.

Correlation of static test results with the NASTRAN stress analysis is yet to be accomplished. Future plans also include a system integration test which will determine heat rejection rates, noise radiation, and structural stability.

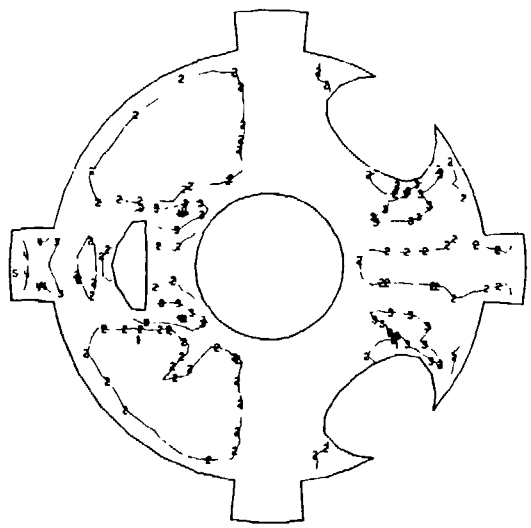

Figure 25 ISO-Stress Contours

Conclusions

The load distribution in a high contact ratio gear design can be adequately calculated using either a static or dynamic analysis for a given profile modification and error condition. The design can be further enhanced by including the buttress tooth feature and modifying the profiles to give essentially a constant relative radius of curvature in the tooth contact to reduce the scoring hazard.

The fabricated stainless steel housing offers a viable alternative to the conventional magnesium alloy casting particularly where high temperature capability and crash survivability are design requirements.

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DISCUSSION

H.Saravanmutto, Ca

Could you comment on the recuperated configuration where the heat exchanger is located between the gas generator and power turbine? Concern is that temperature level at this point would be much higher than that previously used.

Author's Reply

To place the recuperator upstream of the power turbine is a rather new consideration. However, no studies have been made as yet in Germany to evaluate this arrangement. The description of Figure 13 of my paper states that in this arrangement the recuperator is bypassed during full and emergency power. This is done to protect the recuperator from temperatures hotter than in the other installation. Your comments on the heat exchanger bypass is contained in my paper under 'disadvantages'.

DESIGN CRITERIA OF THE A 129 HELICOPTER DRIVE SYSTEM

by

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SUMMARY

This paper discusses the design philosophy of the Agusta A 129 drive system which is actually under development to meet the modern requirements of the Armed Forces for a light helicopter in the anti-tank role with night and day fighting capability.

Efforts have been made to meet the stringent requirements of system layout and to achieve low weight, high life, maximum reliability, survivability and ballistic tolerance, through integrated system design, modular concepts, use of redundant system and dry run capability.

INTRODUCTION

The transmission system discussed in this paper was designed for the A 129, helicopter now under development at Agusta, in conformity to the Italian Army requirements for a light anti-tank helicopter.

The most stringent requirements for the helicopter are :

- twin turboshaft power
- inherent provisions inside the mast for the installation of a night and day visionics system
- transportability
- ballistic tolerance

and for the dynamic system :

- 2500 hours min. T.B.O.
- 30 minute safe flight capability after loss of lubricant
- accessories ground run capability with main rotor at rest.

In addition to the above requirements, are also requested:

- ease of maintenance
- high reliability
- good weight/power ratio

DESIGN PHILOSOPHY

An extensive and laborious redesigning of the main trasmission lay-out was required during the preliminary study phase for a rational integration of the transmission into the helicopter, because of the limited space provided for the dynamic system and the adoption of unusual requirements.

An easy maintenability is an essential feature in a military helicopter to achieve maximum system reliability and thus minimize mission abortions.

Experience however teaches that "maintenance generates maintenance" and accordingly the reliability of a mechanical system is even lower when life limited items are involved. Starting from the fact that some parts may fail, the introduction of modularization criteria for the subsystems of a primary system such as the main transmission, can make maintenance easier, allowing the replacement of high failure risk modules and then eliminate the possibility of adversely affecting reliability, for the greater the separation between the modules the lower the risk of secondary damage.

These criteria have been complied with during design, with the result of achieving not only a satisfactory accessibility to the main drive system and engine modules, but also to the accessories and hydraulic components located in the same area.

From this viewpoint the main transmission has been partitioned into four partially in-

dipendent subsystems with possibility of disassembly directly on the helicopter.

The modules are :

1) Input quill first reduction stage and actuated free wheeling,
2) Main gear case housing the II, III, IV reduction stages,
3) Upper case with main rotor mast and bearings,
4) Accessories gear box.

In addition to modularization the possibility has been retained of indipendent removal of traditional maintenance components such as free-wheels and lubrication pumps.

The lubrication system was designed featuring two modules: the first with the cooling fan straight connected to the accessories gear box; the second integrating the cooler body, filters, bypass thermostatic valve, pressure relief valve, safety and check valve, with no need for external oil line and fittings.

Besides the unusual but recurrent external limitations, to the A 129 transmission was imposed an internal limitation as well, constituted by the large diameter of the mast.

As it is possible to see from the transmission scheme, the mast diameter has not allowed to achieve a great reduction ratio for the planetary stage, because large diameters proved necessary for the sun gear, the shafts and the bearings coaxial with the mast itself.

In order to overcome these limitations, the transmission ratio of 78.2 was obtained through four reduction stages tailored in such a way as not to interfere with the ring diameter of the planetary stage, which on the other hand, was required to provide a rational suspension system for the transmission.

The selection of the four reduction stages was positively evaluated during the preliminary design phase, because it proved convenient even with respect to weight optimization.

The use of spiral bevel gears for the first two stages has allowed to free the power transmission scheme from engine center distance and mounting attitude, and to contain the collector gear diameter (third reduction stage) within the planetary ring gear diameter.

Moreover this solution has allowed to position the free-wheels in an area accessible from the outside without removal of the first stage transmission module and allowing for the power line of engine n° 1 only, to incorporate disingagement provisions for the free-wheel, as necessary for ground running the accessories with the main rotor at rest.

With the advent of stringent military requirements for attack helicopter survivability, several design vulnerability aspects have been explored, and among them of course the transmission system scheme, which for the A 129 has been designed to feature duplication of the rotor-engine power line up to the end reduction stage (planetary).

The modular transmission lubrication system has been designed to duplicate the most vulnerable conmponents keeping them as far as possible apart with no need for connecting lines.

Also the main transmission suspension system has been designed with vulnerability rationale through the adoption of eight large diameter rods, convergent into duplex junctions, which have already been experimented with success on the A 109.

Vulnerability criteria were given paramount importance in the design of tail rotor shafting. The choice of a supercritical system was primarily dictated by the requirement for large diameter shafts which not only had to feature ballistic tolerance but also had to guarantee torque transmission following a hit.

The weight increase generated by the use of oversized diameter shafts with respect to the normally transmitted torque, is partially recovered by the suppression of the hangers necessary for a subcritical drive systems, thus minimizing the number of couplings and bearings, which generally also reduce system reliability.

An intermediate or tail rotor gear box, lubricated by oil, may be adequately designed to provide required failsafe operation as well as to meet vulnerability requirements, one of the basic advantages of the oil lubricated gearbox is the possibility to place an electric chip detector in the best location to monitor metal chips generated by gears or bearings failure. An oil temperature sensor may also be easily installed.

However seal leakages and complete oil loss problems, have suggested the use of grease lubrication for these gearboxes.

This kind of lubrication requires gearboxes of particular internal architecture such that the grease migrates within the cavities around the parts to be lubricated, thus providing several compartements which prevent that grease be completely drained in case of ballistic casing damage.

POWER TRANSMISSION SCHEME

As mentioned above the main transmission is a four stage reduction system.
The first and second stages consist of spiral bevel gears, the third stage of helical gears which drive the final gear consisting of a spur gears planetary system.
The tail rotor shaft is driven directly by the collector gear and by a spiral bevel gears set.
The accessory gearbox can be driven both by the collector gear (in flight) and by the gear shaft of the left engine first stage (on ground) and has the possibility to drive one alternator , a hydraulic pump, an ECS filter compressor and the lubrication oil cooling fan.
The power transmission scheme is completed by two engine shafts provided with flexible couplings and by a supercritical shafts system which drive the intermediate and the tail-rotor gearboxes.

CONFIGURATION AND WEIGHT TRADE-OFFS

Because of its functional features the A 129 transmission does not fit into a normal weight comparison diagram versus the ratings integrated by transmission ratios.
As a matter of fact its weight is greatly affected by the mast dimension which houses a part of the stationary and rotating flight controls and by the accessory gearbox which can be driven both in flight and on ground with rotor at rest without the aid of an APU.
However in the design definition phase all the esperimented technological features necessary to minimize weight have been applied to each component without altering the component functional features.

DESIGN CRITERIA

CONDITION	ENGINE			MAIN ROTOR OUTPUT		TAIL ROTOR OUTPUT		
	RPM	▲ SINGLE HP	▼ TWIN HP	RPM	HP	RPM	TRUST kg	HP
STATIC LIMIT	27000	1341	2196	346	1982	1657	708	466
TRANSIENT	--	--	--	--	--		596	371
2 1/2 MINUTES	--	--	--	--	--		460	254
60 MINUTES	27000	894	--	--	--		374	182
MAXIMUM CONTINUOUS		816	1464	346	1318		345	140
100% POWER RATING		732	1464	346	1318		345	140

▲ based on sea level, ISA standard day

▼ based on hovering at 2000 m , 35°C, with factor of 1.20

GEAR DESIGN CRITERIA

The first two reduction stages consist of spiral bevel gear sets. Being the pitch line velocity of both gears very high (> 10000 ft/min), the gears have been designed featuring limited dynamic loads and thus minimize the probability of scoring and bending overstressing. To achieve the required dynamic load condition, fine diametral pitches and high mismatch contact ratios have been adopted in conjunction with low design bending stresses; surface roughness has been limited to 16 CLA max. to assure a good tribological behaviour. The small teeth size has implied verification of the top land thickness in order to avoid problems of overcarburization.

The third reduction stage consists of two helical pinions driving a collector gear. In this reduction stage the choice of helical teeth was a trade-off between weight, face contact ratio and axial thrust; moreover a fine diametral pitch allowed a thin rim with relevant weight saving.

To provide a smooth run and consequently low noise level a face contact ratio larger than 2.0 was imposed for both helical and spiral bevel gears.

The final reduction stage is a spur gear epicyclic system for which a high gear ratio could not be achieved because of the stringent dimensions imposed to the sun gear and to the fixed ring. However a gear ratio of about 3.0 for this application was judged satisfactory.

In order to achieve the above gear ratio and at the same time save system weight, a pressure angle of 25° was chosen for the sun-planet mesh and of about 17° for the planet-ring mesh.

To allow a good load sharing and concurrently reduce the effect of the vibrations on the main case, a flexible flanged ring gear was adopted. This solution was already successfully adopted on the A 109 main transmission.

As mentioned above the accessory gearbox has the peculiarity of being driven in two different ways, which implies that each gear can be driven or driving.

For the drive train of this gearbox were selected spur gears, expressly designed to avoid "scoring" in both running conditions.

The shape and the stiffness of all main transmission fast gears will be analitically dimensioned by means of a computer program capable of predicting their natural frequencies which shall be later experimentally measured.

BEARING DESIGN CRITERIA

The L_{10} life for all bearings has been calculated with the aid of a computer program for high speed bearings, which in the statistical determination of life takes into account the internal loads generated by centrifugal forces and by gyroscopic moments.

The bearing internal geometry has been devoted careful attention to offset the effects resulting from rolling element dynamics; in fact a sudden change in ball control at such high speeds can cause catastrophic failures because of the quick wear caused by friction temperature rise.

Besides, high speed bearings failures have such a high progression rate that they cannot be timely detected by conventional diagnostic systems.

For these reasons high speed bearings have been designed with a higher than average reliability parameter and their housings have been provided with temperature bulbs to detect any sudden rise which is an indication of malfunction.

All roller bearings have inner races worked integrally with the shaft to avoid inner race creeping on the respective shafts.

The reliability of all bearings has been further increased by the use of M50 VIM-VAR steel for the rolling elements and rings and by the use of silver plated steel cages. These materials are essential to overcome the high temperatures during the oil off run condition.

Both inner races of the mast bearings are also worked integrally with the shaft made from ESR 4340 (H_{RC} 53÷55), thus avoiding large diameter nuts.

FREEWHEELS

Sprag type clutches are installed in the first stage spiral bevel gear designed so that they engage centrifugally. This feature ensures a continuous contact between sprag and race and ensures smooth reingagements at speed, after a period of freewheeling.

A good tradeoff between weight and cost has encouraged the selection of this kind of clutch.

The shaft of input N° 1 is axially displaced so as to prevent freewheel engagement

and thus independently drive the accessories gearbox on ground.

Further, in order to avoid the risk of axially loading the sprag, the spiral bevel set was designed to take only radial loads on the gear.

LUBRICATION SYSTEM

The lubrication and cooling system consists of two volumetrical gerotor pumps which suck from different sumps and feed two separate lines provided with relief valves and filters.

The two systems are integrated with sump - mounted mechanical safety valves, eccessive differential pressure sensor, and fluid-mechanical valves which cut off the oil flow if a damage exists on the line.

Cooling is provided by a single fan which supplies the air flow to the two indipendent radiators which too incorporate thermal bypass valve.

Extreme care was exercised to avoid the presence of external lines and fittings and to make the oil jets of the lubrication line inside the transmission, accessible for inspection.

In addition, suitable provisions have been developed to ensure safe oil off running condition, such as servicing the gears with impregnable material and collection sumps having the function to gradually return the oil for a short time, in default of normal lubrication.

DIAGNOSTICS

The A 129 is provided with an Integrated Multiplex System capable of processing the data fed by the sensors from the critical points of the drive system and of making them available to the pilot continuously or on demand.

As said above, the tail rotor gearboxes are grease lubricated, and they need accelerometers as detecting equipment.

The main transmission is oil lubricated and is provided with the normal instrumentation to survey pressure and temperature. It is also provided with a Quantitative Debris Monitoring (Q.D.M.) system to detect the magnetic chips generated by the failure of bearings, gears, etc.

Thermal probes integrate both the tail rotor gearbox accelerometers and the first stage QDM system of the main transmission.

VULNERABILITY

One of the most important requirement of the Italian Army is that the helicopter be able to fly for at least 30 minutes if hit by a 12.7 caliber bullet.

The whole transmission system of the A 129 was designed reflecting this requirement and a great care was given to the most exposed areas in such a manner that all gears were straddle mounted, bearings were protected by ESR 4340 steel liners of adequate thickness and hardness; the lubrication system were the most redundant as possible; and the rotating parts could function for at least 30 minutes without lubrication.

The temperature rise due to oil off running was taken into account dimensioning the gears backlash and the bearings radial play, and selecting no preloaded ball bearings.

The tail rotor supercritical shafts, the grease-lubricated gearboxes and the main rotor shaft which have the function to protect the rotating controls, were also designed reflecting vulnerability criteria.

CONCLUSIONS

Respondence to the imposed requirements has been achieved through integration of the transmission system into the helicopter, principally through the adoption of design rationale and confining new technologies to those components, which because of their nature, could not be conventionally developed.

During components design development, preliminary tests shall concurrently be conducted in order to substantiate the choices made in the course of the preliminary study phase and thus limit risk areas in the course of assembly development tests.

APPENDIX A

A 129 DESCRIPTION

The Agusta A 129 is a light, twin turboshaft powered, combat helicopter under development for the Italian Army use, primarly in the anti-tank role. It has a single, four-blade, articulated main rotor and a two-blade, semirigid, tail rotor.

The crew of two is seated in tandem with the aircraft commander/pilot seated aft and above the copilot/gunner. Armament is carried on four pylons mounted on stub wings.

The primary armament is the Hughes Aircraft Company TOW missiles system. Rockets, machine guns pods, and external fuel tanks can also be carried on the armament pylons.

Propulsion is provided by two Rolls Royce GEM-2 engines capable of developing up to 918 SHP each.

The take-off gross weight in the Italian Army primary mission configuration is 8080 lbs (3665 kg).

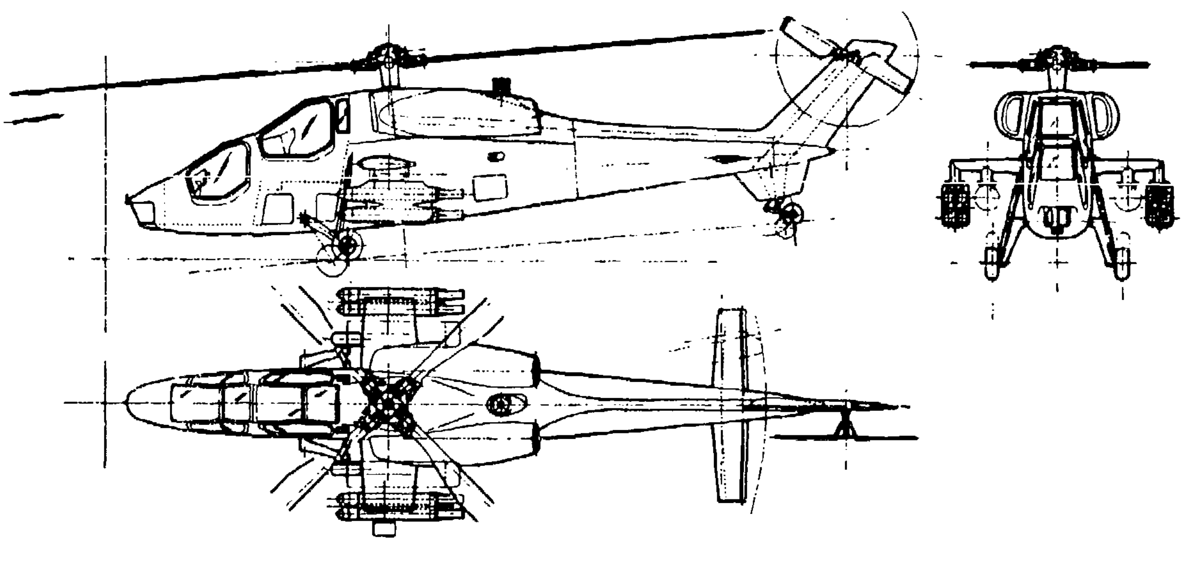
The A 129 has performance comparable to that of the U.S. Army's Advanced Attack Helicopter (AAH) but is considerably smaller and lighter. The AAH, of course, carries twice as many missiles as the A 129.

The A 129 is the first helicopter to protect the upper flight controls from ballistic damage, wire strikes and icing by placing the control rods and the swashplate assembly inside the main rotor shaft. An inner stationary mast provides inherent provisions for installing the stationary or telescoping Mast Mounted Sight (MMS).

The Integrated Multiplex System is the single most significant feature in the A 129. It provides this small helicopter with unprecedented capability and flexibility.

The heart of the IMS is a reduntant MIL-STD-1553B data bus communication and centralized data processing system which totally integrates navigation, communication, flight instruments and controls, system management, aircraft monitoring and maintenance recording, greatly reducing crew work load.

In addition the A 129 is a "tough little machine" which is difficult to detect, difficult to hit and if hit can continue flight for at least 30 minutes, but if it does crash the airframe is crashworthy,and the crew is protected and will survive to the 90th percentive of a survivable accident (as defined by MIL-STD-1290).



AGUSTA A129

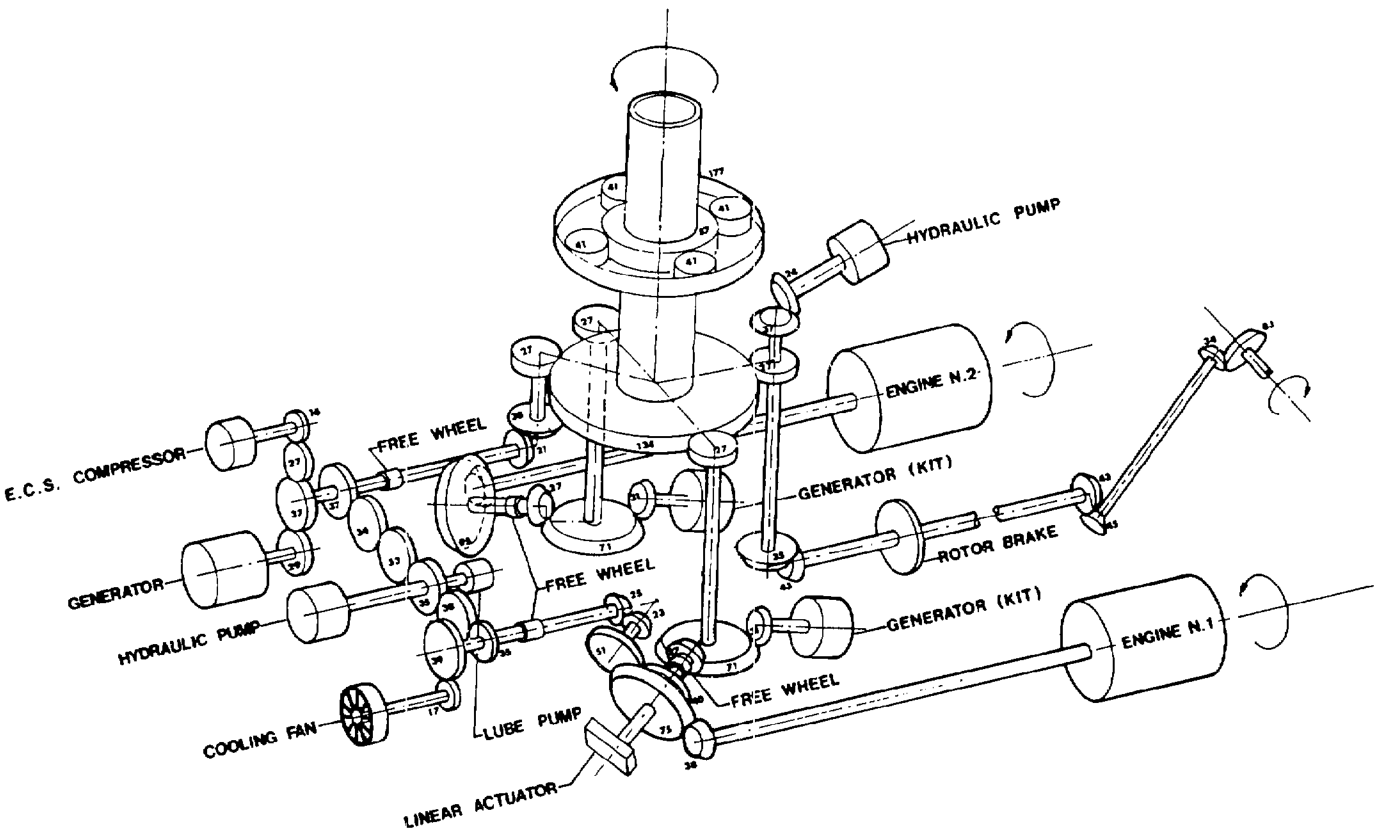


Figure 1. A129 - DRIVE SYSTEM SCHEMATIC.

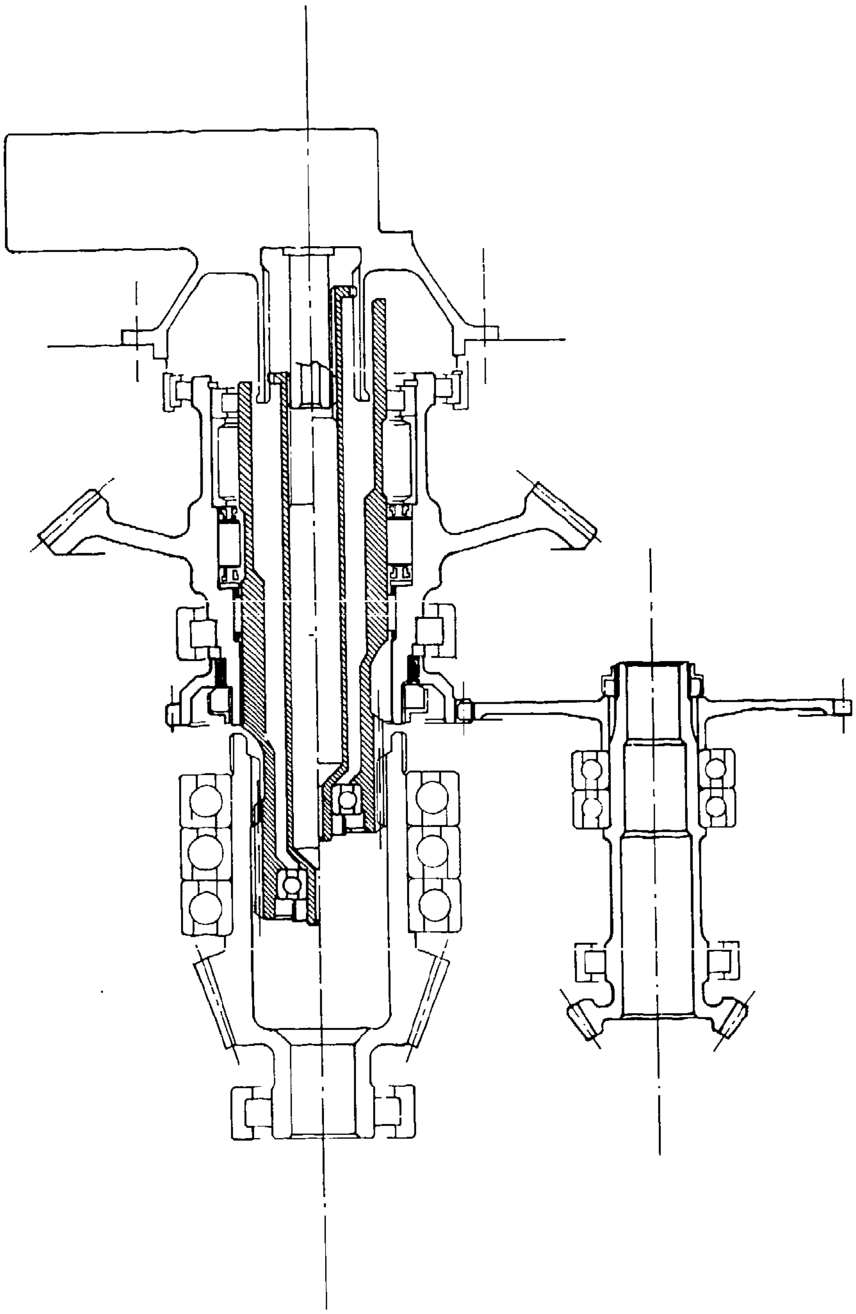


Figure 2. A129 - ACTUATED FREE WHEEL UNIT

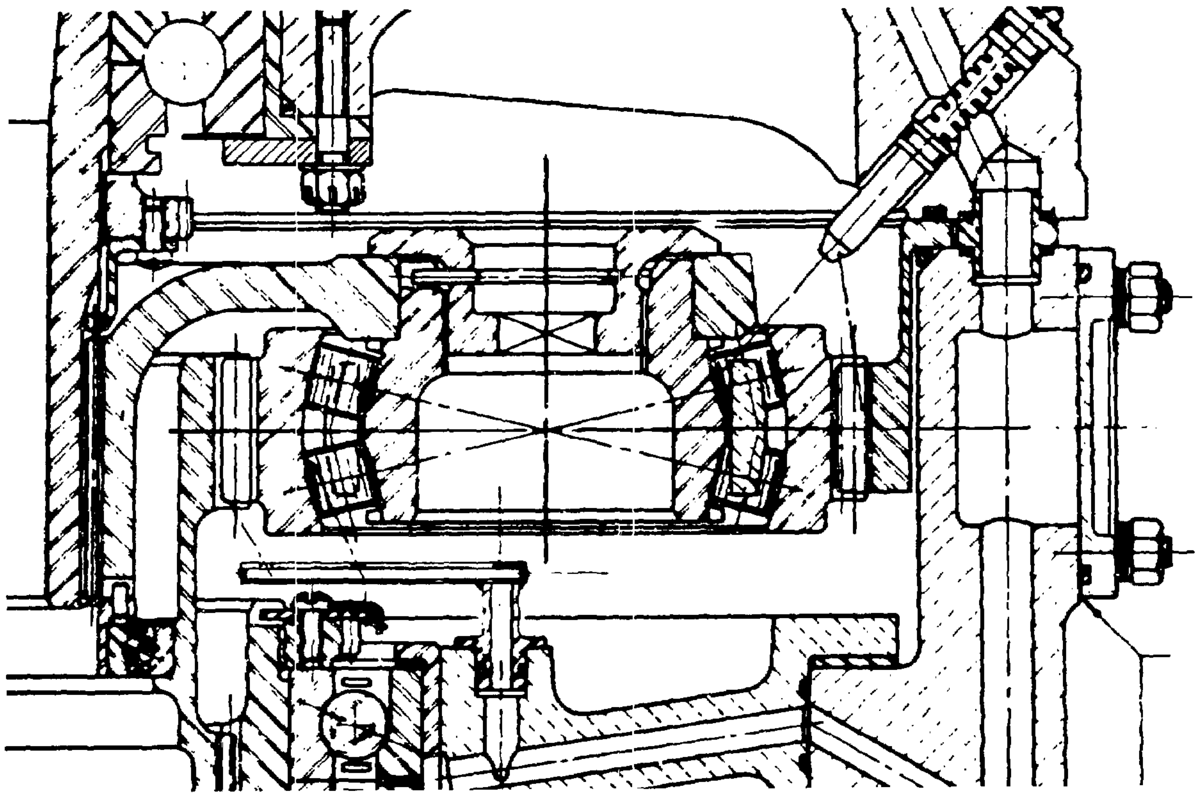


Figure 3. A109 - PLANETARY SYSTEM

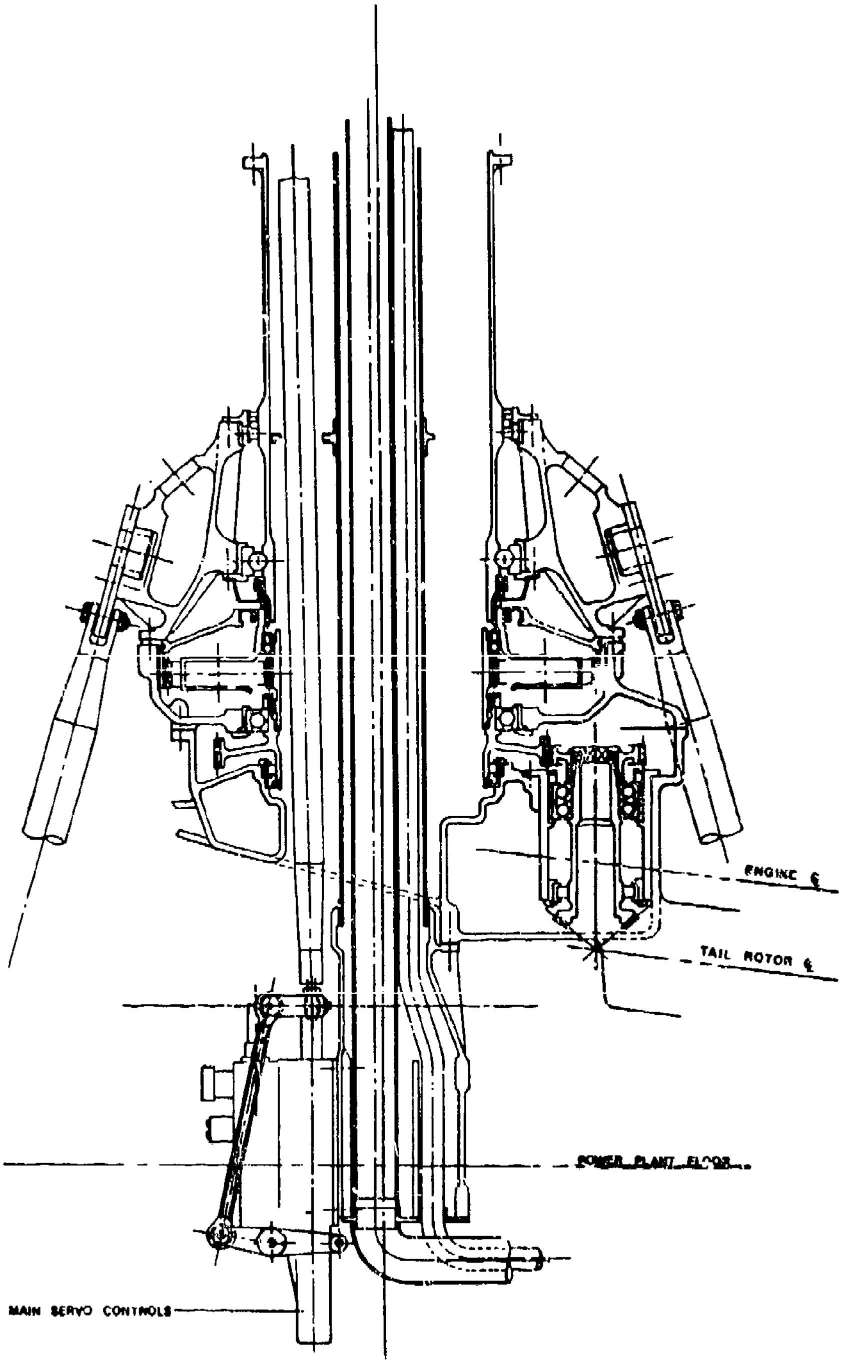


Figure 4. A129-MAIN CASE CUTAWAY VIEW

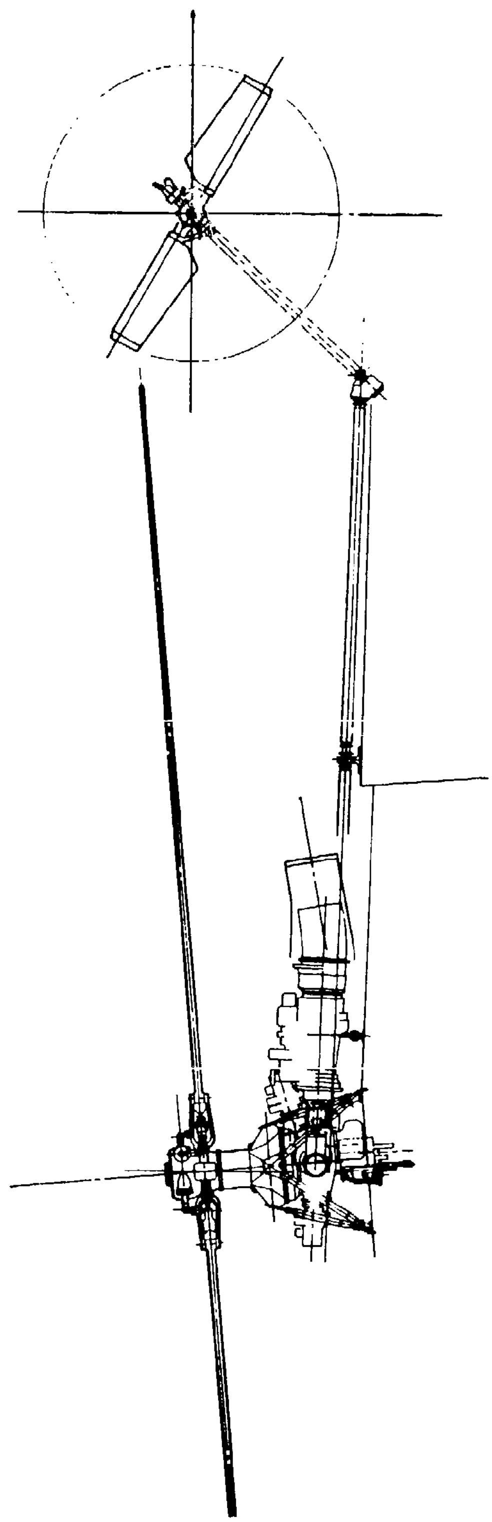


Figure 5. A129 PROPULSION AND DRIVE SYSTEM

DISCUSSION

Unknown Questioner

Have you conducted tests of the air intake shown in your paper to determine whether or not they will accommodate foreign object ingestion?

Author's Reply

Although the shape looks peculiar, tests have been conducted to measure its resistance to foreign object damage. Ice or dust effects are still to be determined.

Same Questioner

How successful have you been with the post-to-carrier attachment system described in your paper?

Author's Reply

No problems have been encountered thus far in the program.

LE DEVELOPPEMENT DES HELICOPTERES EN FRANCE

par
l'Ingénieur en Chef de l'Armement
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75 996 PARIS ARMEES

Les deux dernières decennies ont correspondu, pour l'industrie française des hélicoptères, à une période de croissance rapide : d'une part les forces armées nationales étaient dotées, en quantités significatives, d'appareils militaires bien adaptés à leur finalité opérationnelles et de bonnes qualités techniques, d'autre part des succès considérables étaient obtenus, sur le marché de l'exportation, aussi bien avec les hélicoptères militaires qu'avec d'autres machines, à vocation civile, développées en prenant en compte les impératifs du marché, et se classant souvent de manière très favorable par rapport aux produits de la concurrence.

La présentation qui suit retrace les caractéristiques essentielles du développement des hélicoptères en France durant cette période.

Elle analyse par ailleurs la ligne directrice et les modalités de l'action des services publics dans ce domaine.

Enfin, en définissant les besoins futurs et les axes d'action, elle esquisse les perspectives pour le futur.

1. HISTORIQUE DES DEVELOPPEMENTS D'HELICOPTERES FRANCAIS

La planche 1 rappelle la succession des différents programmes d'hélicoptères français, avec la quantité totale cumulée d'appareils produits par l'industrie nationale. (Les dates indiquées sont celles du premier vol du prototype).

Il est intéressant de situer cette évolution dans le cadre de l'évolution mondiale du produit hélicoptère (planche 2).

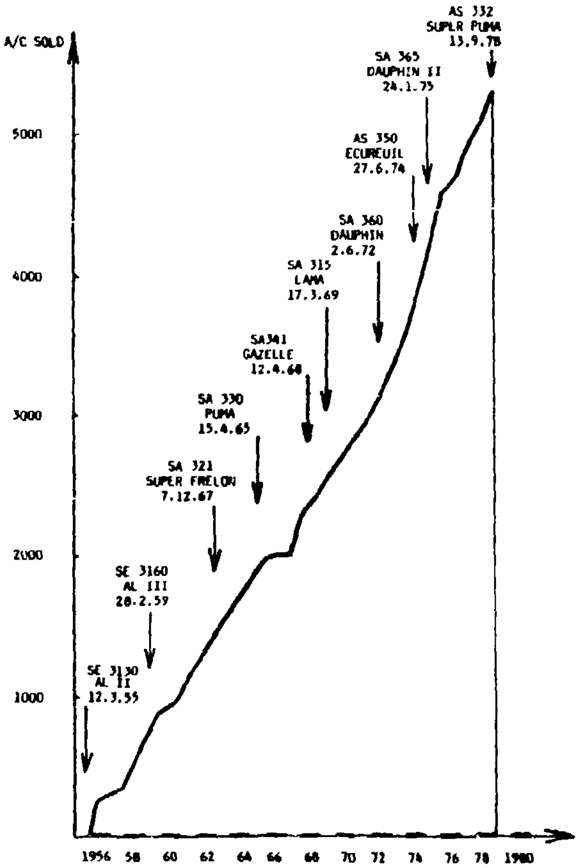

PLANCHE 1

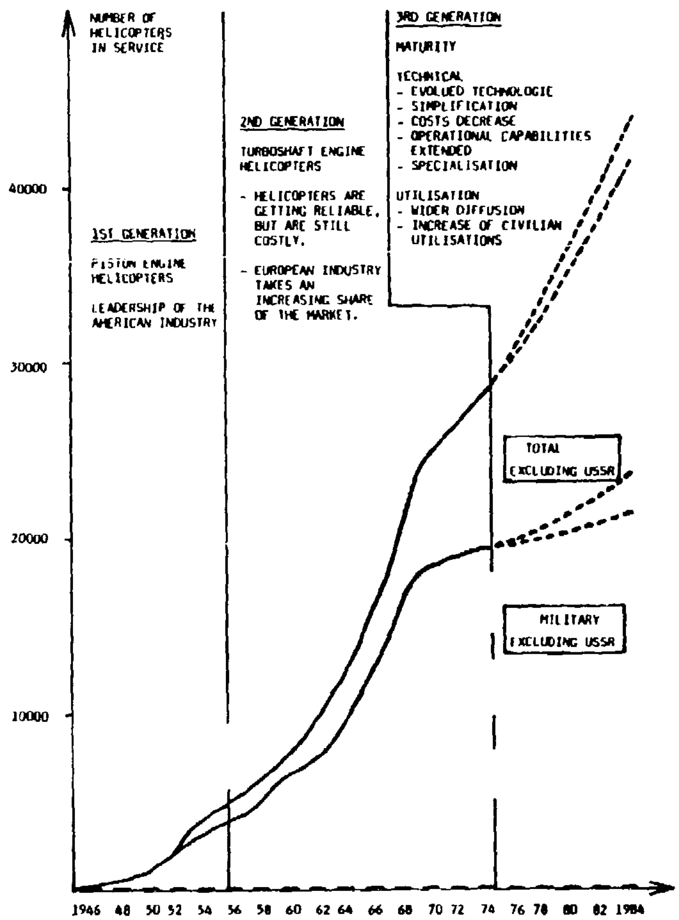

PLANCHE 2

En comparant ces deux planches, l'on constate que l'activité française a suivi cette évolution mondiale, et l'a même, dans certains cas, devancée.

Pour commenter rapidement cet historique, je distinguerai trois catégories

- les appareils des années 1960
- les appareils de la coopération franco-britannique
- les appareils de la nouvelle gamme.

1.1.- Les appareils des années 1960

Il s'agit de la famille des ALOUETTE et du SUPER FRELON.

1.1.1.- Les ALOUETTE

1.1.1.1.- L'ALOUETTE II

Appareil développé en 1955 pour remplir au profit de l'Armée de l'Air, de l'Armée de Terre et de la Marine les missions suivantes :

- liaison
- écolage
- observation - PC volant
- sauvetage
- évacuation sanitaire
- transport de passagers et de fret
- soulèvement de charges.

De plus, une version civile a également été définie.

Les caractéristiques essentielles de ce programme sont données en planche III.

1.1.1.2.- L' ALOUETTE III

Cet appareil n'a pas été développé au titre d'un programme, mais comme extrapolation et modernisation de l'ALOUETTE II, en 1960.

L'appareil était destiné à exécuter avec plus d'aisance les missions dévolues à l'ALOUETTE III, avec en plus :

- pour l'Armée de Terre :
  . la reconnaissance armée
  . le transport de 6 commandos équipés
- pour la Marine :
  . la sauvegarde porte-avions de jour
  . des missions de liaison sur petits bâtiments.

Les caractéristiques de l'ALOUETTE III sont données en planche IV.

PLANCHE III

ALOUETTE II

- Caractéristiques principales

  SE 3130 moteur : ARTOUSTE II B 1
  puis
  ARTOUSTE II C

  SE 318 moteur : ASTAZOU II A

- 5 Sièges
- Masse maximale : 1600 kg
- Vitesse de croisière maximum : 180 km/h
- 1er vol : 1955
- Livraisons (31.12.80) Etat français : 444
  Autres utilisateurs français et exportation : 861

PLANCHE IV

ALOUETTE III

- Caractéristiques principales
- SE 316 O
  361 B Moteur ARTOUSTE II B
  316 C

  7 sièges
  Masse maximaum : 2100 kg
  Vitesse maximum de croisière 190 km/h
- SE 319 moteur ASTAZOU XIV
  Vitesse maximum de croisière 197 km/h
- 1er vol
  - SE 316 1960
  - SE 319 1971
- Livraison

	SE 316	SE 319
Etat français	141	73
Autres utilisateurs français et exportation	1094	111

PLANCHE V

LAMA

- Caractéristiques principales

  SA 315 B Moteur : ARTOUSTE II B
  Masse maximum au décollage : 1950 kg
  2300 kg cargo sling
  Charge maximale à l'élingue : 1000 kg cargo sling
  Vitesse maximale de croisière : 193 km/h
  Plafond à 1950 kg : 5400 m
- Premier vol : 1971
- Livraisons : 290 (31.12.80)

Hélicoptère encore en production.

1.1.1.3.- Le LAMA

Cet appareil a été conçu pour le travail aérien : il est destiné à porter des charges élevées et à travailler à haute altitude.

Il correspond à une cellule d'ALOUETTE II sur laquelle est monté le groupe moto-sustentateur de l'ALOUETTE III.

Ses caractéristiques essentielles sont données en planche V.

1.1.2.- **Le SUPER FRELON**

Cet appareil a été conçu pour répondre à une fiche programme de la Marine de 1962, pour les missions suivantes :

- lutte anti sous-marine
- transport d'assaut
- transport sanitaire
- transport de frêt
- récupération de têtes d'engins.

Il a été développé à partir de 1961.

Ses caractéristiques essentielles figurent sur la planche VI.

PLANCHE VI

SUPER FRELON

SA 321 G

- Caractéristiques principales

SA 321 G	Trois moteurs TURMO III C 5	
	C 6	
	C 7	
Masse maximale	13 000 kg	
Vitesse maximale de croisière	255 km/h	
- Premier vol 1962		
- Livraisons 31.12.80	Marine française	: 24
	Autres utilisateurs français et exportation	: 74

1.2.- Les appareils de la coopération franco-britannique

Il s'agit des appareils couverts, pour le développement et la production, par l'accord intergouvernemental franco-britannique signé en mars 1977, qui prévoyait une organisation désignant, pour chaque programme, une Agence Exécutive nationale et un maître d'oeuvre industriel unique.

Ces hélicoptères sont :

- le SA 330 PUMA
- le SA 341 GAZELLE
- le WG 13 LYNX

1.2.1.- Le SA 330 PUMA

Cet appareil a été conçu pour répondre à la fiche programme de l'Armée de Terre du 17 Mars 1962, et aux caractéristiques militaires précisées le 15 juillet 1963.

Les missions principales étaient les suivantes :

- héliportage d'éléments d'infanterie
- hélitransport en zone avancée
- hélitransport à grande distance.

Les caractéristiques essentielles de l'appareil devaient être :

- aptitude au vol IFR
- mise en oeuvre autonome
- facilité de maintenance
- facilité d'évolutions à très basse altitude.

L'appareil a été choisi en février 1967 par le Royaume-Uni, pour équiper les formations de la RAF. Seules donc son industrialisation et sa production ont été réalisées en coopération, le développement ayant précédé et ayant été effectué nationalement.

Les caractéristiques essentielles du SA 330 sont données planche VII.

1.2.2.- Le SA 341 GAZELLE

Cet appareil a été conçu pour répondre aux exigences communes franco-britanniques d'un hélicoptère léger d'observation. Les missions qui lui sont dévolues sont les suivantes :

- reconnaissance
- liaison
- observation
- PC volant
- support d'armement léger
- évacuation sanitaire
- transport de fret léger.

En outre, l'hélicoptère peut être utilisé en appareil école.

Cet appareil a été développé et produit en coopération SNIAS/WHL, sous agence exécutive et maitrise d'oeuvre française.

Il a fait l'objet, en 1977, d'une revalorisation (SA 342).

Les principales caractéristiques de la GAZELLE sont données planche VIII.

1.2.3.- Le WG 13 LYNX

Cet appareil a été développé en coopération franco-britannique (SNIAS-WHL) sous maitrise d'oeuvre et agence exécutive britannique. Utilisé en version "utility" par l'Army britannique, il a également donné lieu à une version navale pour la Navy britannique et la Marine française. Cette dernière version comportant un système d'armes anti-surface et anti sous-marine (SONAR) dont le développement était effectué par la France.

Les caractéristiques essentielles du LYNX version Marine française sont données en planche IX.

PLANCHE VII

SA 330 PUMA

- Caractéristiques principales
- SA 330 Moteurs 2 TURMO III C 4
- 18 passagers
  ou
  12 commandos
- Masse maximale 6,7 T
  7 T
  7,4 T
- Vitesse maximale de croisière 258 km/h
- Premier vol : 1965
- Livraison (31/12/80)
  - Etat français : 173
  - Etat britannique : 44
  - Autres utilisateurs français et exportation : 440

PLANCHE VIII

GAZELLE

- Caractéristiques principales
  - SA 341 Moteur ASTAZOU III N
    - 5 sièges
    - Masse maximum 1800 kg
    - Vitesse de croisière maximum 240 km/h
  - SA 342 Moteur ASTAZOU XIV
    - 5 sièges
    - Masse maximum 1900 kg
    - Vitesse maxumum de croisière 264 km/h
- Premier vol
  - SA 341 1968
  - SA 342 1978
- Livraisons (31.12.80)
  - Etat français : 189
  - Etat britannique : 243
  - Autres utilisateurs français et export : 414

PLANCHE IX

LYNX MK 2

- Caractéristiques principales
  Deux moteurs ROOLS ROYCE BS 360 GEM 2 900 SHP (671 kW)
  - Masse maximale : 4422 kg
  - Vitesse de croisière maximale : 140 knots (259 km/h) B = O ISA + 15° C
  - Missions principales

  A - Détection et attaque de sous-marins (ASW)
  B - Recherche de surface par radar
  C - Attaque de bateaux de surface
- 1er vol : 1978
- Livraisons (31.12.80) Marine française : 26 Marine britannique : 55 Armée britannique : 78 Exportation : 41

1.3.- Les appareils de la nouvelle gamme

Ce sont les appareils qui constituent, actuellement, l'essentiel de la production nationale française, et qui sont représentatifs de l'état de la technologie nationale. Il s'agit des appareils de la famille DAUPHIN, des ECUREUILS et du SUPER PUMA.

1.3.1.- Les appareils de la famille DAUPHIN

Le DAUPHIN a été initialement développé comme le successeur de l'ALOUETTE III, avec le DAUPHIN SA 360, de 2800 kg. Cependant, la demande du marché se précisant, une version plus lourde et bimotorisée a été développée (SA 365 C) puis une version de performances améliorées a vu le jour (SA 365 N).

C'est ce dernier appareil, qui incorpore les caractéristiques les plus avancées de la technologie française (moyeu rotor et pales en matériaux composites, rotor AR fenestron, etc...) qui constitue le cheval de bataille de la production nationale dans le créneau des bimoteurs moyens, et qui a remporté plusieurs succès importants à l'exportation.

Les principales caractéristiques des appareils de la famille DAUPHIN sont données planche X.

1.3.2.- Les appareils de la famille ECUREUIL

Ces appareils ont été développés pour créer, dans la gamme des hélicoptères légers, des produits susceptibles de prendre la suite des ALOUETTES II et III, de caractéristiques modernes et attractifs vis à vis de la concurrence.

Développé initialement en version monomoteur (avec moteur américain pour le marché de l'Amérique du Nord) l'ECUREUIL est le résultat d'un effort particulièrement poussé d'analyse de la valeur.

Une version bimoteur, destinée elle aussi au marché américain, a été réalisée, et a rencontré un succès rapide.

Les caractéristiques des appareils ECUREUIL sont données planche XI.

1.3.3.- Le SUPER PUMA

Cet appareil a été développé pour donner un successeur au PUMA, avec un produit plus moderne, amélioré, et susceptible de soutenir la concurrence vis à vis de l'UTTAS des Etats-Unis.

Il s'agit d'une revalorisation du PUMA, dont il reprend la forme de fuselage, mais en réalité cet appareil est le résultat d'un effort considérable de développement, qui a concerné la structure (amélioration des caractéristiques anticrash) les ensembles mécaniques et la motorisation (revalorisation des performances par adoption de turbines MAKILA nouvelles) qui a été guidé par le souci d'améliorer la facilité et l'économie de mise en oeuvre de la machine.

Les principales caractéristiques du SUPER PUMA sont données planche XII.

PLANCHE X

DAUPHIN

- Caractéristiques essentielles
  - SA 360 C Moteur : 1 ASTAZOU XVIII A
  - SA 365 C Moteur : 2 ARRIEL 1 B
  - SA 365 N Moteur : 2 ARRIEL 1 C
- 13 passagers
- Masse maximum SA 360 C 3 T
  SA 365 C 3,5 T
  SA 365 N 3,8 T
- Vitesse de croisière : SA 360 C 270 km/h
  SA 365 C 252 km/h
  SA 365 N 293 km/h
- Premier vol - SA 360 1972
  SA 365 C 1975
  SA 365 N 1979
- Livraisons SA 360 30 SA 365 65

PLANCHE XI

ECUREUIL

- Caractéristiques principales
  - AS 350 Moteur (1 ARRIEL 1 B
    (1 LYCOMING LTS 101
  - AS 355 Moteur 2 ALLISSON C 20 F
  - 5 passagers
  - Masse maximum AS 350 1950 kg
    AS 355 2100 kg
  - Vitesse maximale AS 350 272 km/h
    AS 355 240 km/h
  - Premier vol AS 350 : 1974
    AS 355 : 1979
  - Livraisons 405

PLANCHE XII

SUPER PUMA
- Caractéristiques principales
- AS 332 Moteurs : 2 MAKILA
- 21 passagers
- Masse totale : 7800 km
- Vitesse maximum de 286 km/h
- Premier vol 1978
- Livraisons 2
- Commandes 51

2. CARACTERISTIQUES DES DEVELOPPEMENT D'HELICOPTERES FRANCAIS

Les caractéristiques essentielles qui se dégagent de cette revue du développement des hélicoptères en France sont à mon avis les suivantes :

Premièrement, l'on constate que l'effort de développement des hélicoptères en France a permis à l'industrie nationale de proposer une gamme très étendue d'appareils. Actuellement encore, alors que la production d'ALOUETTE II est terminée, l'Aérospatiale est sans doute l'hélicoptériste mondial qui propose la gamme de produits la plus étendue. Cette vaste gamme, qui exclut cependant les extrêmités, c'est à dire l'hélicoptère très léger ou l'hélicoptère très lourd, est le résultat d'une politique délibérée de l'industrie et des services publics. Outre un intérêt commercial évident, elle présente l'avantage de confronter l'industrie nationale à l'ensemble des problèmes techniques posés par l'hélicoptère, et ceci constitue un élément stimulant pour des équipes techniques et industrielles, qui ainsi peuvent se tenir prêtes à aborder dans de bonnes conditions et avec expérience le développement d'un appareil, dans n'importe quel créneau pour lesquelles se dessineraient des perspectives commerciales intéressantes.

Cette politique a été rendue possible par la vitalité du motoriste national, qui avait su acquérir dans les années 1950 une avance certaine dans le domaine des petites turbomachines (l'ALOUETTE II a été le premier hélicoptère à turbine), et qui a su développer son activité vers les turbines de plus grande puissance.

Deuxièmement, l'on observe, sur l'historique des développements d'hélicoptères français, certaines périodicités intéressantes :

D'abord une période d'environ 3 ans entre l'apparition de nouveaux modèles : cette périodicité est très importante pour le maintien du potentiel technique de la branche : en effet, dans un domaine à l'évolution technique rapide, la compétitivité s'incarne dans une équipe de techniciens capables d'innovation. Il est impossible de maintenir cette équipe "dans la course" sans un exercice permanent, qui est constitué justement par la sortie de prototypes régulièrement échelonnés. Cette périodicité de 2 à 3 ans correspond à la durée de l'étude d'un prototype, et le fait qu'elle ait pu être respectée (en particulier dans les années 1960 grâce à la continuité des programmes militaires officiels), a constitué sans aucun doute un élément décisif des succès passés et actuels.

Ensuite, la période de vie d'un appareil (entre le début de l'étude prototype et l'obsolescence technique commerciale ou opérationnelle) pouvant être chiffrée à une vingtaine d'années, l'on s'est efforcé de procéder à des revalorisations de l'appareil à mi-vie, en montant un moteur plus performant, et en appliquant de nouvelles techniques .

<u>Exemples</u> : AL II ARTOUSTE 1951
AL II ASTAZOU 1962
AL III ARTOUSTE 1960
AL III ASTAZOU 1970
PUMA T III CA 1969
SUPER PUMA (MAKILA) 1980.

Cette politique, qui maintient la compétitivité des produits et rentabilise au maximum les efforts de développement, a été pratiquée en France avec bonheur, et a été encouragée par les services publics.

3. MODALITES DE L'ACTION DES SERVICES PUBLICS

Les résultats obtenus par la France dans le passé pour les hélicoptères peuvent être considérés comme flatteurs. Ils sont dus bien sur d'abord au dynamisme de l'industrie nationale cellule et moteurs, mais aussi, dans une mesure importante, à l'action des services publics. Cette action a revêtu des formes variées.

En premier lieu, les armées nationales ont financé intégralement les études et le développement d'un certain nombre de programmes (AL II, PUMA, SUPER-FRELON, GAZELLE) et assuré le débouché industriel initial de ces programmes: on a vu que dans les années 1960, la continuité et la fréquence des besoins militaires avait constitué un élément très important de stimulation, qui a largement concourru à créer dans l'industrie une équipe de haute compétitivité.

Mais ce qui était possible dans un contexte de croissance rapide du secteur hélicoptère et alors que les forces armées accédaient à l'hélicoptère, est moins facile dans ces dernières années, alors que le marché militaire national se trouve équipé et relativement saturé par des produits dont la longévité a augmenté, et dès lors que les développements se sont relativement renchéris.

D'autre part, dans les dernières années, l'industrie nationale a résolument attaqué le marché civil, avec des produits qui lui étaient adaptés.

Dans ces conditions, les Services publics ont cherché à mettre en oeuvre d'autres moyens d'action pour favoriser et guider l'effort de développement des hélicoptères en France. C'est ainsi que s'est développée une politique d'avances remboursables consenties par le Ministère des Transports et gérées par le Ministère de la Défense, et d'avances remboursables consenties par le Ministère de l'Economie. Ces avances ne correspondant qu'à une partie de l'effort de développement industriel, se caractérisent par des conditions de remboursement avantageuses, et amènent l'Etat à partager pour une part, le risque initial de développement et d'industrialisation des programmes. Les programmes DAUPHIN, ECUREUIL et celui du moteur ARRIEL ont bénéficié de telles avances. Il est à noter toutefois que ce système présente évidemment, par rapport au système de financement des développement par le budget des armées, l'inconvénient commercial de grever le prix des matériels produits du montant des remboursements dus.

Enfin, des formules de soutien mixtes ont pu être envisagées : c'est en particulier le cas pour le SUPER PUMA et son moteur MAKILA, appareil à vocation polyvalente militaire et civile, dont le développement a été soutenu à hauteur de 65 % par l'Etat sous la forme d'un financement budgétaire militaire, d'une avance du Ministère des Transports et d'une avance du Ministère de l'Economie.

Troisièmement, les pouvoirs publics se sont préoccupés de soutenir et d'orienter l'action industrielle dans le domaine de la préparation de l'avenir.

C'est ainsi qu'un programme de recherches et d'études, concernant l'ensemble des disciplines de l'hélicoptère est mené dans l'industrie et dans les centres officiels. Ce programme, d'un volume croissant, correspond pour la part effectuée par l'Aérospatiale en 1981 à plus de 2,5 % du chiffre d'affaire de cette société, et est soutenu par l'Etat pour plus de la moitié, l'autre part étant autofinancée par l'industrie. C'est dans les actions menées au titre de ce programme qu'il faut rechercher l'origine de certains éléments qui ont constitué, après développement complémentaire au titre des programmes, des atouts des hélicoptères français de la nouvelle gamme (pales plastiques, éléments de structure en matériaux composites, moyeux en matériaux composites, rotors AR de type fenestron).

4. PERSPECTIVES POUR L'AVENIR

Le marché mondial de l'hélicoptère est en expansion à un taux de l'ordre de 6 à 8 %, (alors que la tendance d'accroissement du parc militaire, dans ce marché est plus faible, de l'ordre de 1,5 à 2,5 %). Dans ces conditions, et compte tenu de la position qu'elle occupe actuellement, la France envisage l'avenir dans le domaine des hélicoptères avec dynamisme.

En ce qui concerne les programmes militaires futurs, les besoins prévisibles sont les suivants :

- un hélicoptère antichar de 2ème génération, capable du tir de nuit et doté si possible d'une capacité d'appui-protection, d'une masse inférieure à quatre tonnes, et devant entrer en service à la fin des années 1980. La France et la RFA examinent actuellement la faisabilité d'un tel programme en coopération, dans le cadre des délais et des montants budgétaires impartis

- un hélicoptère de transport tactique de masse moyenne, assurant la relève des PUMA et SUPER PUMA, et présentant par rapport à ces appareils des avantages opérationnels et de coût d'entretien significatifs, pour une mise en service au début des années 1990, et dont devra dériver une version marine

- un hélicoptère léger d'observation et de liaison, pour remplacer dans le courant des années 1990 les ALOUETTES obsolescentes.

Pour le marché civil, il faudra prévoir, dans la logique de la gamme de produits actuels :

- des améliorations de l'ECUREUIL, pour maintenir sa compétitivité sur le marché mondial

- la poursuite de la carrière du DAUPHIN, dans le créneau très convoité des bimoteurs moyens, grâce à une revalorisation passant par une motorisation améliorée utilisant une nouvelle turbine aux caractéristiques comparables à celles des moteurs du programme ATDE (revalorisation accompagnée d'une diversification de l'appareil vers les utilisations militaires)

- un successeur, amélioré sur le plan de la technologie et du coût d'opération, des PUMA et SUPER PUMA, à l'horizon 1995

- enfin un hélicoptère pouvant succéder, avec une technologie moderne et des coûts réduits, au LAMA dans son rôle de travail aérien.

Quelles devront être dans ces conditions les légères directives de l'action des services officiels :

- D'abord orienter, de façon générale, les actions d'études et de développement dans le sens de la polyvalence des matériels, de sorte que les besoins civils et militaires puissent être remplis par des matériels présentant le maximum de communalité (ce qui ne veut pas dire qu'il ne sera pas nécessaire, pour tel ou tel besoin militaire ou civil bien identifié, de développer des matériels spécialisés)

- Ensuite, et de façon corollaire, promouvoir au niveau officiel une politique concertée d'association de financements militaires et de concours civils (en provenance du Ministre des Transports et de l'Economie), dont un exemple peut être pris dans la façon dont le programme SUPER PUMA a été soutenu par l'Etat

- Troisièmement, accentuer l'effort étatique et industriel sur les actions de préparation de l'avenir, de sorte à maintenir le niveau technologique du secteur hélicoptère, et à le préparer à vaincre les points durs des développement à venir

- Enfin, promouvoir la coopération européenne. On a vu a quel point la coopération franco-britannique avait eu un effet stimulant et favorable sur l'activité hélicoptère en France . L'extension au niveau européen de cette coopération, qui se justifie par la concomitance des besoins européens sur l'ensemble des programmes militaires futurs, est une nécessité pour le futur, et constitue une chance que nous devrons saisir, ce à quoi nous nous préparons.

ENTREES D'AIR D'HELICOPTERES

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RESUME - Sur les hélicoptères de faible et moyen tonnage, pour des raisons d'architecture de l'appareil, les moteurs sont situés en arrière de la tête rotor, ce qui rend leur alimentation très malaisée : c'est le cas en particulier de l'Ecureuil AS 350 et du Dauphin SA 365 qui sont équipés de moteurs de nouvelle génération ; ces turbomoteurs sont à aspiration axiale ou annulaire suivant que le réducteur est situé du côté de la tuyère ou du compresseur.

Une étude approfondie des entrées d'air dès le stade du projet peut permettre de les concevoir avec un minimum de répercussions néfastes sur la puissance fournie par les moteurs, sur la traînée de l'appareil et les risques de pompage.

De ce fait, une telle étude est susceptible de réduire sensiblement les délais et donc les coûts de mise au point en vol. Cet exposé résume les méthodes utilisées à l'Aérospatiale et les résultats obtenus en vol sur différents appareils.

NOTATIONS

A = Surface
V = Vitesse
P = Pression statique
P_T = Pression totale
ρ = Masse volumique
DC60 = Indice de distorsion

$$= \frac{(\overline{P_{T_2}} \text{ sur un secteur de } 60°) \text{ minimum} - \overline{P_{T_2}}}{\frac{1}{2}\,\rho_2\, V_2^2}$$

INDICES

$\overline{}$ = Valeur moyenne
0 - Infini amont
1 = Entrée d'air
2 = Compresseur
4-5 = Venturi
6 = Sortie

1. INTRODUCTION

Dans une grande partie du domaine altitude-température, les performances des hélicoptères bi-moteurs sont limitées par la puissance disponible plutôt que par la capacité des organes de transmission.

En particulier, pour ceux qui sont soumis à la FAR 29, catégorie A, la panne moteur au décollage est très sévère ; dans le cas du DAUPHIN bi-moteur SA 365, par exemple, 5 % d'écart sur la puissance moteur représentent 180 kg de masse décollable.

Ce chiffre de 5 % correspond à une valeur courante des pertes de puissance à l'avionnage relevées sur les hélicoptères qui n'ont pas fait l'objet d'une étude poussée de l'interface aérodynamique moteur-fuselage et en particulier des entrées d'air.

Cet exposé décrit les problèmes posés par la définition des entrées d'air moteur :

- fonctionnement aérodynamique dans tout le domaine de vol, pertes de puissance, augmentation des consommations, diminution de la marge au pompage des moteurs, décollements sur le fuselage pénalisant l'appareil en traînée,
- adaptation à l'environnement = protection des moteurs à l'ingestion de corps étrangers, sable, givre,
- facilité d'installation dans la structure de l'appareil et maintenance en service.

Les méthodes employées à l'Aérospatiale Marignane sont développées ainsi que les résultats obtenus par une approche systématique sur différents appareils récents comme les DAUPHIN SA 365N, SA 366G Coast Guard, TWINSTAR AS 355 et SUPER PUMA AS 332.

2. PROBLEMES POSES PAR LES ENTREES D'AIR D'HELICOPTERE

2.1. Fonctionnement aérodynamique

Pertes de puissance à l'avionnage

Elles sont définies par les écarts de puissance constatés entre le banc et le vol, à même régime réduit du générateur. Ces pertes sont fréquentes et peuvent provenir :

- d'une réingestion d'air chaud provenant du refroidissement d'huile, du recyclage des gaz d'échappement par le rotor principal ou directement des tuyères moteur dans les configurations de vent arrière. Rappelons que 1°C d'élévation de température moyenne dans l'entrée d'air coûte environ 0,8 % de puissance.
- de pertes de charge ou de perturbations de l'écoulement moyen dans l'entrée d'air : en moyenne, 1 %

(ou 10 mb) de perte de pression totale correspond à au moins 2 % de perte sur la puissance disponible. L'effet des distorsions de pression et de la turbulence est mal connu, mais peut être responsable de pertes inexpliquées par détérioration du rendement compresseur.

Pertes sur les consommations

Ces pertes peuvent être importantes et proviennent d'une détérioration du cycle thermique du moteur.

Elles peuvent être causées par :

- une élévation de température moyenne à l'entrée compresseur
- une diminution du rendement compresseur due aux distorsions de pression.

Augmentation de la traînée appareil

Une surface d'entrée d'air et une lèvre mal adaptée peuvent entraîner des décollements sur les capots responsables d'une augmentation de l'ordre de 10 % de la traînée de l'appareil complet.

Problèmes de pompage, mise au point en vol

Ces problèmes sont les plus délicats parce qu'ils font intervenir le dialogue motoriste-avionneur, sans que, en l'absence de critères et de mesures précises, la responsabilité de l'un ou de l'autre ne puisse être clairement établie.

Les problèmes de pompage par vent arrière ont souvent exigé une mise au point en vol fastidieuse qui aboutit en général à des modifications de tuyères.

Tous ces points ont des répercussions importantes sur les performances de l'appareil, consommations, distances franchissables, masse décollable, vitesse maximum, coût de développement.

2.2. Adaptation aux conditions de vol

Le bon fonctionnement du moteur dans tout le domaine de vol est à démontrer par l'avionneur pour obtenir la certification de l'appareil. Dans ce but, il faut procéder à des essais d'ingestion de corps étrangers (oiseaux, grêlons, etc...) et des essais de givrage.

Sur les appareils récents, une protection efficace est assurée par une grille (diamètre du fil 0,8 mm - maille de 5,5 mm) montée devant l'entrée d'air éventuellement munie de raidisseurs. Ce dispositif est pénalisant en performances, surtout en vol d'avancement, car il produit une perte de charge et une traînée importante à grande vitesse.

Certaines conditions d'utilisation exigent même l'emploi d'un filtre anti-sable constitué d'un grand nombre de tubes Vortex séparant les particules solides par centrifugation. La perte de puissance et de masse décollable dûe à cet accessoire peut être élevée.

Le spectre de bruit des entrées moteurs est noyé dans le spectre général et ne fait pas l'objet d'études systématiques.

2.3. Installation dans la structure appareil

Les critères suivants peuvent orienter le choix d'une solution :

- facilité de maintenance sous capots
- légèreté
- esthétique, en particulier pour le marché civil, où les entrées d'air simples, de type PITOT, sont difficilement acceptées.

Le dessin des manches à air est particulièrement compliqué dans le cas des moteurs situés en arrière de la tête rotor : il est impossible d'éviter des coudes ou une longueur importante de manche à air. Le souci d'éviter les fuites dans celle-ci, causes de pertes importantes par réingestion de gaz chauds, peut rendre nécessaire l'installation de joints mobiles, autorisant l'ouverture des capots d'accès aux organes mécaniques.

3. MOYENS EXPERIMENTAUX EN VOL

Les essais d'entrées d'air en vol sont très difficiles à analyser et à interpréter tant qu'il n'est pas possible de disposer un peigne comportant un nombre suffisant de prises de pression totales devant le compresseur, et d'effectuer des mesures simultanément sur toutes les sondes au cours du même vol.

Actuellement, sont utilisés une douzaine de capteurs différentiels classiques reliés aux sondes par une certaine longueur de tuyau.

Une étude visant à développer un équipement de mesures instationnaires est en cours.

Les ingestions de température sont estimées à l'aide de sondes à thermocouple.

4. MOYENS EXPERIMENTAUX EN SOUFFLERIE

Ce sont essentiellement des outils de vérification des hypothèses émises au niveau de la définition : ils permettent l'analyse fine des écoulements et la quantification des performances de l'entrée d'air.

La figure 1 montre le principe de fonctionnement d'une maquette échelle 1/2 construite à l'origine pour l'essai des entrées d'air du Dauphin bi-moteur SA 365C.

En réglant la vitesse du vent dans la veine soufflerie, l'attitude de la maquette et de débit des ventilateurs d'aspiration interne, il est possible de simuler le fonctionnement de l'entrée d'air moteur sur pratiquement tout le domaine de vol défini par les paramètres vitesse, incidence, dérapage, altitude-densité, débit masse du moteur.

Etude de l'écoulement interne

Pour chaque cas de vol simulé, un sondage est effectué par le peigne tournant équipé de sondes de pressions totales et par les sondes statiques disposées à la paroi. On obtient :

- la perte de charge moyenne

- l'indice de distorsion

Les capteurs de pression sont montés derrière un scanivalve relié aux sondes de pression par une longueur de tuyau. Avec la chaîne de mesure utilisée, pour l'instant cette installation ne permet pas de mesures instationnaires précises mais l'"épaisseur" du signal relevé donne des indications sur le niveau de turbulence de l'écoulement au niveau des entrées moteur et permet de classer sur ce critère les différentes configurations de maquettes.

Des mesures de température peuvent être effectuées à l'aide de thermocouples placés sur les sondes du peigne de mesure : en simulant une source d'air chaud sur la maquette, les sondages de température permettent d'apprécier la quantité d'air chaud ingéré.

Mesure des traînées

La maquette échelle 1/2 aspire et éjecte le flux interne de simulation des débits moteur à l'intérieur de la veine de la soufflerie : l'ensemble est pesé pour effectuer des mesures comparatives de traînée. Le schéma est conforme à la figure 1.

L'écart de pression mesuré par le venturi entre les sections 4 et 5 donne le débit.

La poussée due au dispositif d'aspiration, projetée sur l'axe de la maquette est par définition la différence entre les flux dynalpiques $\vec{G} = p\vec{A} + \rho V^2 \vec{A}$ entrant et sortant,

$$T = G_6 - G_0$$

qui peut s'écrire

$$T = (G_6 - G_1) + (G_1 - G_0)$$

Le terme $G_6 - G_1$ représente la résultante des actions de la veine d'air comprise entre les sections (1) et (6) sur l'ensemble manches internes, ventilateurs, tuyères.

Le terme $G_1 - G_0$ représente la poussée externe théorique exercée par l'aspiration sur la carène.

La trainée externe de l'entrée d'air est la différence entre cette poussée théorique et la poussée effective T_c que peut réaliser cette carène changée de signe :

$$Tox = - \left[(G_1 - G_0) - T_c\right]$$

Pour pouvoir comparer 2 entrées d'air, en se plaçant à même débit moteur, de manière à respecter le coefficient de débit imposé par le cas de vol simulé, il faut réaliser la même dynalpie de sortie

$$G_6 = P_6 A_6 + \rho_6 A_6 V_6^2$$

La condition $\rho_6 V_6^2 A_6$ est réalisée en maintenant

$$\Delta p_{venturi} = \rho_5 V_5^2 - \rho_4 V_4^2 \text{ constant}$$

La condition $P_6 A_6$ = constante n'est pas réalisable simultanément de manière rigoureuse avec les ventilateurs utilisés et demande une légère correction. Pratiquement, on établit les caractéristiques de traînée en fonction du coefficient de débit en maintenant constant l'écart de pression mesuré au venturi étalonné et en faisant varier la vitesse du vent. On effectue en chaque point la correction de pression du jet.

Etude de l'écoulement externe

L'étude des recyclages d'air chaud par le rotor principal n'est possible qu'à une échelle beaucoup plus réduite, 1/7e ou 1/10e suivant les appareils. Ce type d'essais permet d'analyser les trajectoires de gaz chauds dans les configurations de vol critiques, comme celle de la figure 2 pour le SUPER PUMA, AS 332.

A cause de la taille des maquettes, il est très difficile de simuler directement le champ de température. Les jets chauds sont simulés à froid par injection de gaz carbonique. La mesure des concentrations locales est faite par tubes réactifs et l'application d'une loi de similitude concentrations-températures donne une estimation de la quantité d'air chaud réingéré par les entrées d'air moteur par exemple. Des phénomènes analogues à ceux mentionnés par Boeing lors de l'étude de l'UTTAS ont pu être mis en évidence (référence 1).

5. METHODES D'ETUDE MISES EN OEUVRE

La figure 3 résume les paramètres clés du fonctionnement des entrées d'air sur lesquels il est possible d'intervenir au stade du projet.

Quelle que soit la solution choisie, les performances effectives du moteur en vol dépendent essentiellement, à tuyère donnée :

- du niveau de pression totale moyen devant le compresseur
- du niveau de température totale moyen devant le compresseur.

Les performances du moteur sont mesurées par le constructeur au banc dans les conditions du point fixe de l'hélicoptère, (pression totale à l'entrée = pression statique ambiante) avec un pavillon d'entrée donnant des distributions presque idéales.

Pratiquement, le champ des pressions totales n'est jamais uniforme devant le compresseur : les écarts avec la pression totale moyenne sont caractérisés par un indice de distorsion (écarts dans l'espace) comme le DC 60 et par un taux de fluctuations ou de turbulence par rapport à l'écoulement moyen (écarts dans le temps).

Fonctionnement en vol stationnaire

Le moteur aspire dans tout l'espace autour de l'entrée d'air, mais l'influence de cette aspiration diminue très rapidement avec la distance. La puissance demandée au moteur impose le débit masse et la vitesse V_1. (cf. figure 4)

Le fonctionnement des lèvres peut être expliqué de la manière suivante : les filets d'air proches de la lèvre d'entrée doivent contourner celle-ci et accélérer de la vitesse 0 à la vitesse V_1. La courbure des lignes de courant correspond à une dépression répartie sur la lèvre et dont l'intégrale sur le contour est égal à l'effort d'aspiration.

Plus la lèvre est mince, plus cette dépression est forte et plus le gradient de pression positif imposé par le niveau de dépression moyen dans la manche à air est important. Pour une valeur limite de ce gradient de pression, on atteint le décollement.

Le cas extrême correspond à une lèvre infiniment mince où le décollement est immédiat et où la perte de charge correspond grossièrement à la pression dynamique interne.

Au premier ordre, c'est donc l'épaisseur relative des lèvres qui fixe le coefficient de perte de charge,

$$\frac{\Delta P_T \text{ lèvres}}{\frac{1}{2}\rho_1 V_1^2}$$

et à un moindre degré, le profil des lèvres.

Fonctionnement en vol d'avancement

L'espace peut être séparé en 2 régions (voir figure 5) :

- L'une aspirée par le moteur définissant un tube de courant de surface A_0 à l'infini amont et qui évolue en fonction de la vitesse locale imposée aux filets fluides.

- L'autre éventuellement défléchie mais non aspirée : les filets d'air doivent contourner la lèvre externe en accélérant à partir du point d'arrêt jusqu'à un niveau de dépression fixé par l'épaisseur et le profil de la lèvre (succion) puis en ralentissant pour se raccorder aux conditions imposées par les formes du fuselage. Plus le ralentissement est sévère, plus le décollement risque d'apparaître tôt.

Sur la lèvre interne, le fluide ne subit en général qu'une accélération à partir du point d'arrêt et il n'y a pas de risque de décollement interne.

Le rapport $\boxed{\varepsilon = \frac{A_0}{A_1} = \frac{\rho_1 V_1}{\rho_0 V_0}}$ est appelé coefficient de débit et caractérise l'adaptation de l'entrée d'air.

L'incidence α est définie comme l'angle entre l'axe du tube de courant capté et l'axe de l'entrée d'air.

Ce sont le coefficient de débit et l'incidence de l'entrée d'air qui déterminent l'importance des décollements externes et éventuellement internes sur les lèvres.

Sur la même figure, est schématisé le fonctionnement des entrées d'air statiques (α = 90°) en vol d'avancement : le tube de courant capté n'occupe pas toute la surface de l'entrée d'air, mais seulement une zone "efficace", la partie restante est une zone tourbillonnaire dont l'importance augmente à mesure que le coefficient de débit diminue. Ce tourbillon n'est pas très stable et crée une agitation de la veine fluide dans l'entrée d'air, ce qui se traduit par une fluctuation importante des signaux de pression totale au niveau du compresseur en plus des dégradations de l'écoulement moyen dues au décollement lui-même et au frottement le long des parois.

En vol d'avancement, des pertes de pression totale peuvent provenir des dégradations subies dans le tube de courant, en amont de l'entrée d'air, par frottement le long d'une paroi ou à la traversée d'un obstacle.

6. EVALUATION DES DIFFERENTES SOLUTIONS COMPATIBLES AVEC L'ARCHITECTURE DE L'APPAREIL - CHOIX DE LA POSITION DU PLAN D'ENTREE D'AIR

La figure 6 indique les avantages présumés des différentes solutions d'entrées d'air développées à ce jour sur les hélicoptères où les moteurs sont situés en arrière de la tête rotor.

Ce tableau peut donner des éléments pour le choix de l'implantation du plan d'entrée d'air en fonction de la mission de l'appareil et de l'importance que l'on accorde aux paramètres mentionnés dans la colonne de gauche.

Dans le cas des hélicoptères légers bi-moteurs, performants en vitesse (Bell 222, Sikorsky S76), la solution des entrées d'air latérales semble prédominer actuellement. C'est celle qui a été retenue en finale sur les bi-moteurs Dauphin SA 365 N et Ecureuil AS 355, de préférence au type "frontal" légèrement plus performant, essentiellement en traînée appareil, mais qui présente des difficultés d'installation et une esthétique discutable.

La solution des moteurs montés dans une nacelle "en pod", développée sur des hélicoptères plus lourds, Uttas ou Boeing Vertol Chinook, est très intéressante sur le plan traînée (malgré l'augmentation de la surface mouillée) et performances moteur puisqu'elle permet d'installer une entrée d'air et une tuyère orientées dans l'axe de l'appareil. De plus, cette solution simplifie l'étude des dispositifs destinés à réduire la signature infrarouge des tuyères (applications militaires, montage d'un déviateur de jet). Elle est cependant lourde et peu esthétique.

7. REGLES DE DIMENSIONNEMENT - METHODES DE CALCUL

Section d'entrée d'air A_1 et incidence α

L'optimum du point fixe demande une très grande section d'entrée d'air et est contradictoire avec celui du vol de croisière qui correspond à l'adaptation :

Surface du tube de courant capté A_0 = Surface de la section d'entrée A_1

Le compromis proposé est :

$$\boxed{\frac{A_0}{A_1} = 0,8}$$

A_0 étant calculé pour les conditions du vol de croisière au sol, où ce rapport est proche du minimum, compte-tenu des survitesses locales dues aux formes du fuselage.

Dans ces conditions, une épaisseur de lèvres e de l'ordre de 25 % du diamètre de l'entrée d'air si elle est axisymétrique à incidence nulle α = 0 est le plus souvent suffisante pour éviter les décollements externes et la pénalisation en traînée.

Si l'entrée d'air est en biseau ou accolée à une paroi, l'épaisseur relative de la lèvre diminue puisque c'est la paroi qui devient plan de symétrie et la règle 25 % du diamètre est insuffisante.

Couche limite fuselage

Dans le cas où l'avant-projet fait apparaître l'intérêt d'entrées d'air noyées dans le fuselage, un gain appréciable sur la récupération de pression en vol d'avancement et une diminution très nette de la distorsion peuvent être obtenues en ménageant un piège à couche limite.

Calcul du profil des lèvres et dessin de la manche à air

Pour l'instant seules des méthodes bidimensionnelles sont utilisées, en particulier une méthode semi-empirique utilisant les transformations conformes très rapide à mettre en oeuvre.

Le calcul direct des pressions locales sur une lèvre donnée est effectué par différences finies, d'après un programme ONERA (référence 2). Ce programme est utilisé également pour calculer l'écoulement interne dans la manche à air.

Des méthodes tridimensionnelles sont en cours de mise au point pour cette application à l'ONERA.

La figure 7 montre des lèvres obtenues par la méthode de l'hodographe et adaptées sur le SA 365N DAUPHIN.

La figure 8 montre une comparaison calculs essais sur l'entrée d'air AS 350 Ecureuil au point fixe.

L'écoulement dans la manche à air est accessible par le calcul dans le cas où il n'y a pas de décollement dans la section d'entrée d'air ; des données empiriques comme les tables Data Sheets sont utilisées pour calculer les pertes de charge dues à la manche seule.

Dans les autres cas, en particulier décollés, seul l'essai sur maquette en soufflerie permet une estimation des pertes.

Rappelons que pour une conduite circulaire rectiligne, la perte de charge varie approximativement comme

$$\boxed{\frac{\Delta p_T}{\frac{1}{2}\rho_1 V_1^2} \simeq K\frac{L}{R}}$$

V_1 Vitesse moyenne dans l'entrée d'air imposée par le débit moteur au cas de vol considéré
L Longueur de manche à air
R Rayon moyen
K Coefficient de l'ordre de 0,015 pour une conduite cylindrique

Le choix de la longueur de manche à air résulte d'un compromis, si elle n'est pas dictée par une condition d'architecture générale de l'appareil :

- Une grande longueur, de l'ordre de 8 fois le diamètre compresseur est très favorable pour résorber les distorsions du flux d'air à l'entrée : c'est la longueur nécessaire pour que les grosses structures tourbillonnaires susceptibles d'apparaître dans l'entrée d'air soient détruites.

- Une faible longueur est favorable pour diminuer la perte de charge.

On évite d'avoir recours à des solutions avec volume interne important, formant chambre de tranquillisation, les pertes étant très élevées.

8. RESULTATS OBTENUS

Deux appareils récents sont équipés d'entrées d'air statiques, le DAUPHIN SA 365C bimoteur et l'ECUREUIL AS 350 monomoteur. Ce type d'entrées d'air, on l'a vu présente des avantages sur le plan de la simplicité et de la facilité d'adaptation des dispositifs de protection.

Par contre, à grande vitesse, la distorsion est sensiblement plus élevée, voir figure 9, et les performances moins bonnes que pour les entrées d'air dynamiques, voir figure 10. L'installation d'un filtre en forme de nid d'abeille à grosses mailles dans le plan d'entrée d'air permet de réduire sensiblement le niveau de distorsion et les fluctuations de vitesse devant le compresseur, mais sans apporter de gains significatifs en perte de charge, voir figure 5.

Cette modification a été essayée en vol sur le AS 350 (ARRIEL).

De plus, en 365C, des générateurs de tourbillon ont été placés de part et d'autre de la tête rotor pour limiter les effets du sillage et limiter l'ingestion d'air chaud évacué par le puits BTP : cet artifice permet de rehausser le sillage au-dessus des capots et de dégager une zone d'écoulement sain.

La figure 10 montre également les résultats obtenus en soufflerie pour différentes entrées d'air d'appareils récents. Les entrées d'air dynamiques permettent des performances beaucoup plus intéressantes en vol d'avancement : elles ont été choisies pour les hélicoptères nouveaux DAUPHIN SA 365N, SA 366G COAST GUARD, TWINSTAR AS 355 et SUPER PUMA AS 332.

En SA 365N, en particulier, le plan des entrées d'air a été choisi en avant de la tête rotor de manière à éviter les recyclages d'air chaud.

L'entrée d'air Coast Guard 366G malgré la complication due à l'aspiration annulaire du moteur, ne présente pas de perte sensiblement supérieure à celle du SA 365N.

En AS 355 TWINSTAR, comme les gaz de refroidissement sont canalisés le long de l'arbre de transmission et sont évacués en arrière des moteurs, les problèmes de recyclage d'air chaud en vol d'avancement n'ont pas été rencontrés. Ceci a permis de dessiner des manches à air moins longues, avec des entrées latérales situées au niveau de la tête rotor.

Cette disposition a donné une très bonne efficacité sur tout le domaine de vol.

La figure 11 montre le gain en traînée obtenu par comparaison entre les entrées dynamiques et statiques sur le SA 365N.

Sur le Super-Puma AS 332, un soin particulier apporté au dessin des lèvres et de la grille de protection a permis de gagner 8 % environ sur la traînée totale soit 6 km/h sur la vitesse maximum de l'appareil, par rapport à la première définition.

9. CONCLUSIONS ET RECOMMANDATIONS

Cet exposé résume les méthodes et les résultats obtenus sur les entrées d'air moteur à l'Aérospatiale Marignane. Ces études permettent de limiter de manière très significative les pertes de puissance à l'avionnage, ainsi que les pertes sur les consommations que l'on peut observer sur tous les hélicoptères par comparaison entre les mesures au banc moteur et en vol. De la même manière, il est possible d'éviter des décollements importants sur le fuselage, responsables de pertes de traînée.

Les paramètres essentiels de définition qui conditionnent l'obtention d'un rendement et d'un fonctionnement acceptable sont dans l'ordre d'importance :

- le choix de la position du plan d'entrée d'air sur le fuselage
- la surface de l'entrée d'air
- l'incidence
- le dessin des dispositifs de protection (ingestion de corps étrangers, dégivrage, neige, sable)
- l'épaisseur relative des lèvres
- le dessin de la manche à air
- la forme des lèvres

Les possibilités d'approche par le calcul deviennent de plus en plus importantes mais ne permettent pas encore d'éliminer les essais en soufflerie, si l'on cherche de très bonnes performances.

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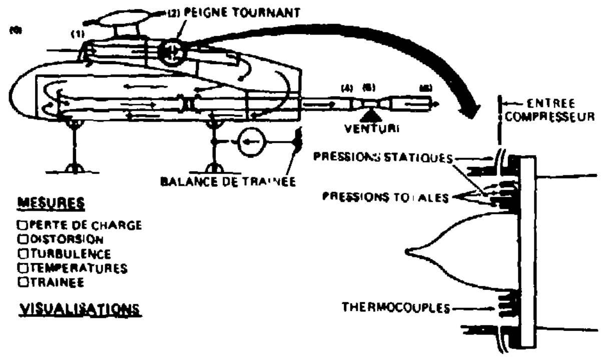


FIGURE 1 : MAQUETTE D'ETUDE DES ENTREES D'AIR DAUPHIN (ECHELLE 1/2)

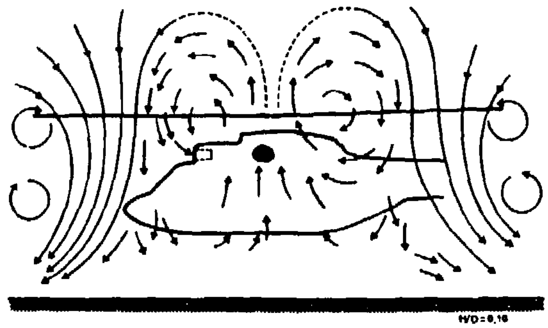


FIGURE 2 : ECOULEMENT AUTOUR D'UN SUPER PUMA EN VOL STATIONNAIRE (DANS L'EFFET DE SOL)

+10° C	DANS L'ENTREE D'AIR COUTENT	8 % DE PUISSANCE
– 1 %	DE PRESSION TOTALE COUTE	2,4 % DE PUISSANCE

PROBLEMES

VOL STATIONNAIRE

- PERTE DE CHARGE DANS LA MANCHE
- REINGESTION DE GAZ CHAUDS

VOL D'AVANCEMENT

- ADAPTATION
- TRAINEE
- INGESTION DE COUCHE LIMITE OU SILLAGE
- REINGESTION DE GAZ CHAUDS
- DISTORSION/FLUCTUATION DEVANT COMPRESSEUR

MOYENS D'ACTION

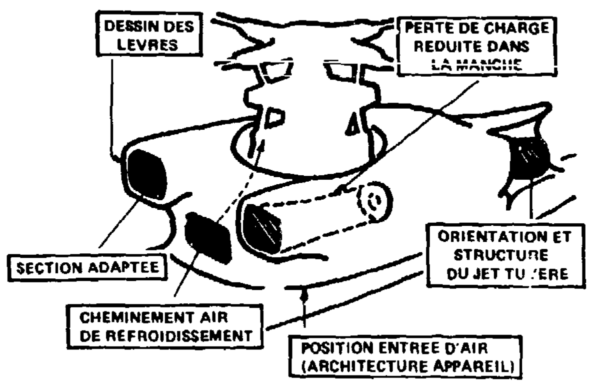


FIGURE 3 : POINTS CLEFS

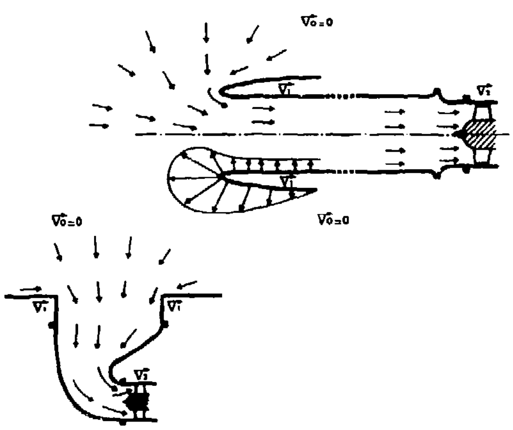


FIGURE 4 : ECOULEMENT DANS L'ENTREE D'AIR EN VOL STATIONNAIRE

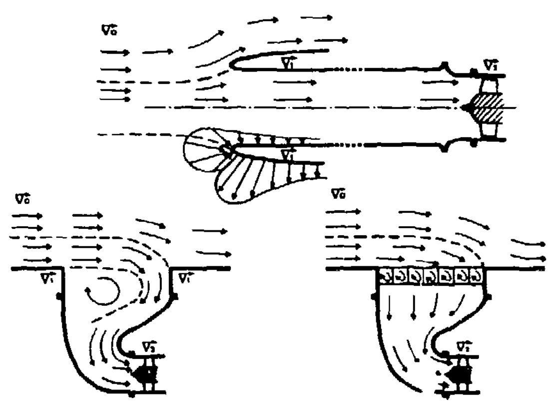
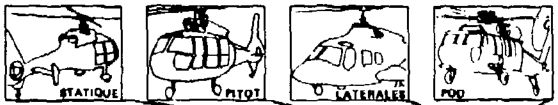


FIGURE 5 : ECOULEMENT DANS L'ENTREE D'AIR EN VOL D'AVANCEMENT



	STATIQUE	PITOT	LATERALES	POD
VOL STATIONNAIRE	●			
VOL D'AVANCEMENT	●	■	▲	■
TRAINEE	●	■	▲	■
ESTHETIQUE	▲	●	■	●
PROTECTION CORPS ETRANGERS	■	▲	▲	▲
FILTRE ANTISABLE	■	●	▲	●
POIDS	■	●	▲	●
MAINTENANCE	■	●	▲	■

● MAUVAIS ▲ ACCEPTABLE ■ BON

FIGURE 6 : COMPARAISON DE DIFFERENTES ENTREES D'AIR

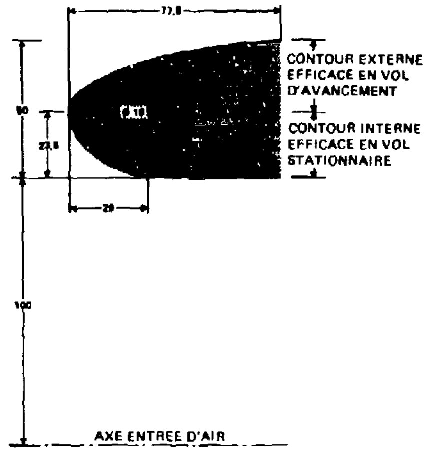


FIGURE 7 : DESSIN DE LEVRE OBTENU PAR LA METHODE DE L'HODOGRAPHE

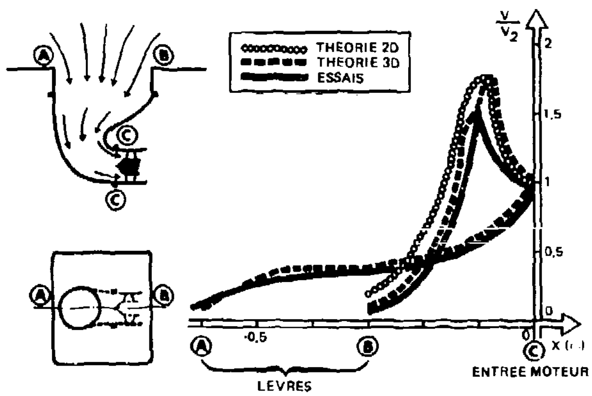


FIGURE 8 : COMPARAISON CALCUL - ESSAIS SUR L'ENTREE D'AIR DE L'ECUREUIL EN VOL STATIONNAIRE

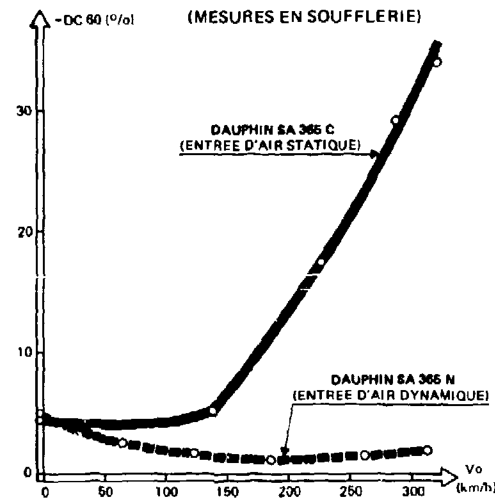


FIGURE 9 : EVOLUTION DE L'INDICE DE DISTORSION EN FONCTION DE LA VITESSE

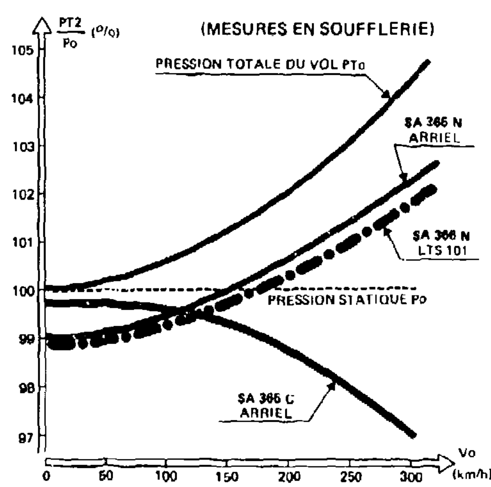


FIGURE 10 : RECUPERATION DE PRESSION EN FONCTION DE LA VITESSE (RAPPORTEE A LA PRESSION TOTALE A Vo = 0)

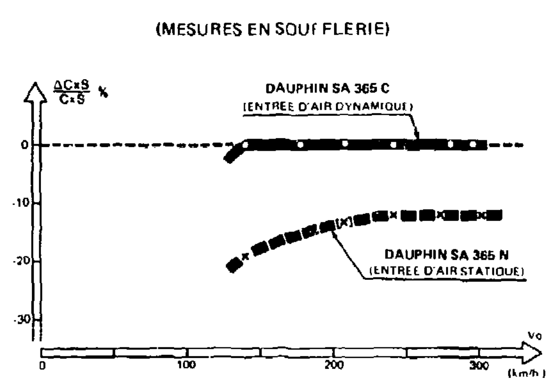


FIGURE 11 : COMPARAISON DE TRAINEE

DISCUSSION

K.Rosen, US

When you modified the SA365 inlet, were there any significant penalties paid in the area of anti-icing as compared with the original static design?

Author's Reply

It is agreed that the problems were greater in the dynamic system than the static system. However, the problems related to this system for our application have been solved.

K.Rosen, US

With respect to your vortex tube inlet, have they worked satisfactorily in relation to icing conditions?

Author's Reply

This question concerns the SA 330 PUMA. The air inlet with vortex tubes has obtained certification after tests conducted in the icing tunnel at the grid (Ottawa, Canada) and in flight under natural icing conditions.

INTAKE DESIGN WITH PARTICULAR REFERENCE TO ICE PROTECTION AND PARTICLE SEPARATORS

By

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and

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SUMMARY

Total environmental protection for helicopter engines is a desirable aim which has not yet been achieved with a reasonable performance penalty. The paper outlines the problems associated with different environments and describes intake systems which have been used to give protection against particular environments. From this information, three intake systems are proposed for further consideration and evaluation in natural environments.

INTRODUCTION

Over thirty years of operation has seen the helicopter used in all the different varieties of climatic zones that exist world-wide. The general adoption of gas turbine engines as the means of propulsion has meant that every deployment of the helicopter to a new climatic zone has brought a new problem to be faced concerning engine intake protection. Development of the gas turbine engine has produced higher compressor speeds and smaller compressor blades which have proved less resistant to damage from external particles ingested by the engine. Therefore, the need for environmental protection has increased over the years.

Role fit intakes have been successfully developed to protect engines from particular environments. However, the role fit philosophy is not attractive to the customer who has a cost, storage, maintenance and operational penalty as a result. Consequently significant development work has taken place over the last decade to produce a basic intake design which has total environmental protection with a minimum of engine and aircraft performance degradation.

THE ENVIRONMENT

One of the major difficulties in protection system design has been the lack of knowledge concerning the actual environment; only when it is accurately defined can definitive rig development work be undertaken. In order to gather the necessary data, progressive evaluation of both the protection system and the environmental measuring system have been required, concurrently, on numerous aircraft flight trials. This has resulted in very slow and expensive progress in the design and clearance of intake protection systems. However, the acceptance test criteria now used for intake protection systems have been shown, by service use, to be adequate in most cases. These test criteria are discussed, briefly, below.

i) Icing Conditions

The presence of water in the atmosphere at temperatures below the freezing point of water can cause ice to form on the air intake and on the forward parts of the engine and the compressor blades. If these parts are not anti-iced, the performance of the engine will be progressively eroded. Subsequently, the ice which has formed will shed into the engine with the probability of damage to the compressor and/or engine surge and flame-out.

Water droplets and ice particles can be experienced in flight both discretely and together (ie in Mixed Conditions). The problems experienced to date have resulted in a multitude of test conditions as illustrated in Table One (proposed UK military requirement). Unfortunately, we are unable at present to rig test conditions such as freezing rain and we are unable to accurately measure snow conditions that occur on environmental trials. The National Gas Turbine Establishment (NGTE) in England is hoping to be able to simulate mixed condtions for WG34/EH101 rig icing trials we shall be conducting later this year.

TABLE ONE

ICING CONDITIONS

Condition	Air Temp $^{\circ}C$	Water Content g/m^3	Duration of Condition	Droplet Size Median Vol Dia Microns	Altitude Range x 1000 ft
I Continuous Maximum Icing	+5 0 -10 -20	0.90 0.80 0.60 0.30	Continuous	20	4 to 10
II Periodic Maximum Icing	+5 0 -10 -20	1.35 1.20 0.90 0.45	15 mins	20	4 to 10
III Mixed Conditions (Continuous)	0 -10 -20	(0.20 LWC (0.60 ICE (0.15 LWC (0.45 ICE (0.10 LWC (0.20 ICE	Continuous		0 to 10
IV Mixed Conditions (Periodic)	0 -10 -20	(0.30 LWC (0.90 ICE (0.20 LWC (0.70 ICE (0.15 LWC (0.30 ICE	15 mins		0 to 10
V Falling Snow (Continuous)	+3 to -20	0.8	Continuous		0 to 10

TABLE ONE (Cont)

Condition	Air Temp °C	Water Content g/m^3	Duration of Condition	Droplet Size Median Vol Dia Microns	Altitude Range x 1000 ft
VI Falling Snow (Periodic)	+3 to -20	1.5	15 mins		0 to 10
VII Recirculating Snow	0 to -20	1.5	15 mins		IGE Hover
VIII Freezing Fog	0 to -20	0.3	5 mins	10 to 20	0 to 50 AGL
IX Freezing Drizzle	0 to -15	0.3 to 0	5 mins	200	0 to 5000
X Freezing Rain	0 to -10	0.3	5 mins	1500	0 to 5000

ii) Sand and Dust

The operation of gas turbine engines in high sand and dust concentrations can cause rapid erosion of the compressor blades and its casing with resultant engine performance and handling degradation.

Various test dusts are in use in the aircraft industry. In the United Kingdom, dust to BS1701 is the general standard used for helicopter engines. This dust has a particle size distribution as shown in Table Two.

TABLE TWO

Particle Dia (Microns)	% weight smaller than dia
20	25
40	41
60	55
80	68
100	79
120	88
140	97
160	100

Intake protection systems which separate more than 85% by weight of this dust from a dust concentration of 16.10^{-4} Kg/m^3 of air have been shown to give acceptable engine life in arid countries.

iii) Salt Water Spray

Once again, various salt spray tests are used worldwide. A typical salt spray solution, for corrosion testing, would have a concentration of 2.10^{-7} by weight of salt in air. In service, the rapid engine performance degradation due to compressor contamination by salt water must be caused by salt concentration levels well in excess of the above figure. However, momentum separation systems for use in icing and sandy conditions have been found to be beneficial in protecting engines from the effects of salt water contamination.

iv) Foreign Objects

Specification foreign object damage testing for engine/intake combinations is generally limited to single tests with small birds and either 25 mm or 12 mm diameter hailstones. However, there is now an increasing amount of ad hoc testing carried out using items which are known to have caused engine compressor damage in-service eg lockwire, nuts, bolts, electrical ties, stones, twigs, ice particles of all shapes and sizes, etc. It is anticipated that aircraft or engine specifications will eventually include more severe engine FOD protection requirements to cover the above items.

OPERATIONAL PROBLEMS

The helicopter is essentially a low level aircraft which normally operates below 3,000 metres. If icing conditions are encountered (ie ice crystals or water droplets below 0°C), the helicopter is often unable to 'climb over' the weather and has to plough on through or turn back. Obviously the latter is highly undesirable in both military and commercial operation and some form of engine intake anti-icing is essential to permit flight in forecast or actual icing conditions. Figure one shows a simple anti-iced pitot intake which is basic fit for the Sea King helicopter.

After having devised a suitable engine intake anti-icing system to permit prolonged immersion in an icing condition, the manufacturer is then faced with the problem of protecting the engine from ice particles which form on and shed from the aircraft fuselage. Typical areas of ice accretion are windscreen support frames and windscreen wipers - as shown in figure two; taken during icing rig trials of a Sea King at the National Gas Turbine Establishment, (NGTE), Pyestock.

The helicopter's major asset is its VTOL ability. However, in doing so, it will generate its own snow or sand and dust storm when operating from unprepared strips or in arid areas where it is impossible to keep landing areas clean. Operating in sandy areas can reduce engine life to tens of hours without an efficient engine intake separation system - especially when several helicopters are operating together. Figures three and four show the Westland Lynx helicopter during environmental trials generating its own sand and snow storms.

A common role of the modern day helicopter is to carry out Search and Rescue operations over the sea - often in gale force winds and breaking waves which cause a high concentration of salt water in the vicinity of a hovering helicopter. Such an environment, illustrated by figure five, can cause compressor fouling within minutes to eliminate the hover power margin of the aircraft. Also, the increasing usage of helicopters from small ships and offshore oil rigs is likely to increase engine salt corrosion problems in the future.

Finally, it is also desirable to protect the engine from debris which finds its way into the air intake during maintenance operations and from various types of debris, other than sand and dust, which becomes airborne when the helicopter is operating near the ground.

EXPERIENCE

In the early days, the problem of intake protection was approached on a role fit aircraft intake basis - sand extractors for arid areas and heated intakes/intake shields for colder climates. Attempts were later made to adopt sand extraction systems for use in icing/snow conditions but these modifications have usually been only of limited use. To illustrate this point, the development of intakes for the Westland Sea King/Commando, the Lynx and the current Sea King Replacement project is reviewed:-

Sea King/Commando

i) The Westland Sea King and Commando are licence-built derivatives of the Sikorsky S61 which was designed with an electrically anti-iced pitot intake (ref figure one). This intake is simple to manufacture, has good aerodynamic characteristics and has a relatively low electrical power requirement. However, early flight trials in icing conditions (back in the 19?0's) showed that large quantities of ice can quickly form on the front of the aircraft, especially around the heated windscreens (ref figure two). When this ice sheds, there is a high probability of it being ingested by the engines - as has been the case on several occasions with resultant engine damage.

ii) The first attempt made to improve the engine icing clearance for the Royal Navy was by means of an electrically heated 'Mushroom Intake' (see figure six). This intake was designed to provide 'line of sight' engine protection from airframe shed ice particles. Following initial rig icing trials at NGTE, this intake was flown in natural icing trials by A and AEE, Boscombe Down. These trials showed that the new intake design induced ice/snow to form on the cabin roof which became an engine hazard: also the electrical power requirement was found to exceed that of the standard aircraft generators.

iii) After the experiments with the 'Mushroom Intake', natural icing trials were conducted with the Sikorsky 'Foreign Object Deflector' or 'Barn Door' intake. Although these trials were successful, concern was expressed over the proximity to the intake of ice which formed on the front of the Barn Door, on the pitot tubes and on the engine bay doors. To remove these problems, a TKS fluid strip dispenser system was added to the front of the Barn Door, as shown in figure seven, and electrical heater mats added in the other areas. System development took place once more at NGTE and further natural icing trials have resulted in an engine icing release which exceeds that of the unprotected rotor system.

The Barn Door intake was designed to provide line of sight protection from airframe shed ice particles. In addition, its shape also provides momentum separation of water droplets from the engine airflow which has been shown, by service use, to be beneficial in reducing the rate of power loss due to engine compressor salt contamination in marine environments.

The fitment of this type of intake to the Sea King produces airframe drag, weight and engine performance penalties: the latter due to both pressure loss and an increase in airflow distortion.

iv) During the same period, Westlands have developed a role fit sand filter pack based on the well known 'Centrisep' tube for Middle East customers of the 'Commando' version of the S61. This pack has also been tested in the NGTE icing facilities and showed that, although forward facing panels quickly block with ice, large side facing panels can provide engine protection from ice particles and water droplets for limited periods (see figure eight - momentum separation of a large part of the water droplets occurring). However, we have reservations on the use of such devices in all types of icing conditions for long periods of time. Also, side facing packs have large intake pressure drops (relative to a pitot intake) which are increased with partial ice blockage giving a large performance penalty due to weight and SFC effects. The inlet distortion characteristics of such packs are normally good and performance penalties reduced if the pack can be by-passed in normal flight conditions - at the expense of further weight, cost and complexity.

v) Westlands have also designed a side facing, electrically anti-iced intake for the Sea King (see figure nine). The object of the design was to maintain the line of sight ice protection and the water droplet momentum separation of the barn door intake whilst improving the airframe drag and intake pressure loss. Wind tunnel model and NGTE icing testing showed the design objects had been achieved, although some further development may have been required to improve the distortion characteristics.

Anti-icing power levels were in excess of those required for the standard pitot intake but less than those for the Mushroom Intake.

Lynx

Similar intake concepts to those used for the Sea King were applied to the design of role fit intakes for the Westland/Aerospatiale Lynx (figure ten) and it's 'big brother' the WG30. The basic aircraft features electrically heated air intakes (see figure eleven) which are sited to give 'line-of-sight' protection from the major areas of airframe ice accretion. Wire mesh debris guards are available for use in areas of high FOD risk and, for hotter climates, we have developed a forward facing sand filter pack as shown in figure twelve.

Sea King Replacement

Total intake protection is required for the Sea King Replacement (WG34). This project, now known as the EH101, is currently being studied by Westlands and Agusta.

We believe that the most difficult intake protection problem is to provide long exposure in icing conditions with minimum basic aircraft performance penalty. Momentum separation of water droplets in both icing and salt water environments has been shown to be essential to reduce the penalty due to the protection system.

The momentum separation principle (as used in the 'Centrisep' and 'Donaldson' tube) is generally accepted as the best means of providing both sand filtration and total FOD protection. This principle has been studied by most of the engine manufacturers in the world and General Electric are the first in the field with an integrated, high performance, co-axial separator on the front of their T700 family of engines - as shown in figure thirteen.

A further use of the momentum separation principle has been used for many years, particularly on Pratt and Whitney PT6 engines. This asymmetric or bypass duct type of device has been extensively developed by Rolls Royce, Leavesden, illustrated by figure fourteen and, by the inclusion of a large bypass duct scavenge flow, they have produced excellent aerodynamic and protection model test results. However, once again, we have our reservations on the adoption of this type of intake. We believe it has many detail design engineering problems in providing the necessary ice free ducting and associated large scavenge flow with the general constraints of a typical engine installation. Also, the random 'bounce' of large ice particles ingested by the intake could cause unacceptable engine damage rates when continually exposed to icing conditions.

After reviewing the above information, a sideways facing anti-iced intake in conjunction with an engine mounted separator was selected as the means of protection for the EH101 (previously known as the WG34). This intake system was chosen as being the lowest risk means of achieving line of sight protection from airframe shed ice, up to five hours mission time in icing conditions and general sand, FOD and salt water spray protection.

Development work on the chosen system is well advanced. Wind tunnel model tests were aimed at producing an intake with low pressure loss in both hover and forward flight, low distortion and low intake entry velocities (to further minimise the risk of ice ingestion) whilst maintaining a shape which will provide good water droplet separation in forward flight. The aerodynamic aims have been met (minimal hover power loss, less than a 100 lb fuel penalty for a five hour mission and acceptable distortion predicted) and an icing rig is being prepared to evaluate anti-icing requirements at NGTE late 1981.

The performance of the General Electric T700 separator is well documented (see Reference 1). Test work to ensure that the sideways facing intake does not compromise the performance of the separator has also commenced with ice ingestion trials at the Lucas facility, Artington. A T700 separator was tested with and without the WHL intake by injecting a large number of different shaped ice particles into and past the intake. The T700 separator had excellent performance with ice particles less than 12 mm cubed and its performance was maintained with the side intake fitted. Particles larger than 12 mm showed random separation thus emphasising the need for line of sight protection for long icing exposure times. Air intake velocities for the side intake were so low that only very small ice particles were attracted towards the intake at hover power conditions.

CONCLUSION

For total environmental protection of the engine air intake as basic aircraft fit, three options are available for further investigation:

a) side facing anti-iced intakes in conjunction with an engine mounted co-axial separator.

b) a bypass intake duct with anti-icing and a large scavenge flow.

c) large side-facing vortex tube type [illegible].

Each of the above intake protection systems has its own penalties of weight, drag, engine power and SFC, cost and manufacturing problems. In order to minimise the penalties, close co-operation is required between airframe and engine manufacturers throughout the design and development stages of the helicopter.

Adoption of any of these three options will probably give an engine protection standard which will permit sufficient exposure times in hostile environments such that engine power and integrity is no longer the limiting factor; only further natural environmental trials will yield the answer!

REFERENCES

1 T700 Engine Integral Inlet Separator - All Weather Operation
By M G Ray and J L Browne, General Electric Company

Fig.1 Sea King Pitot intakes

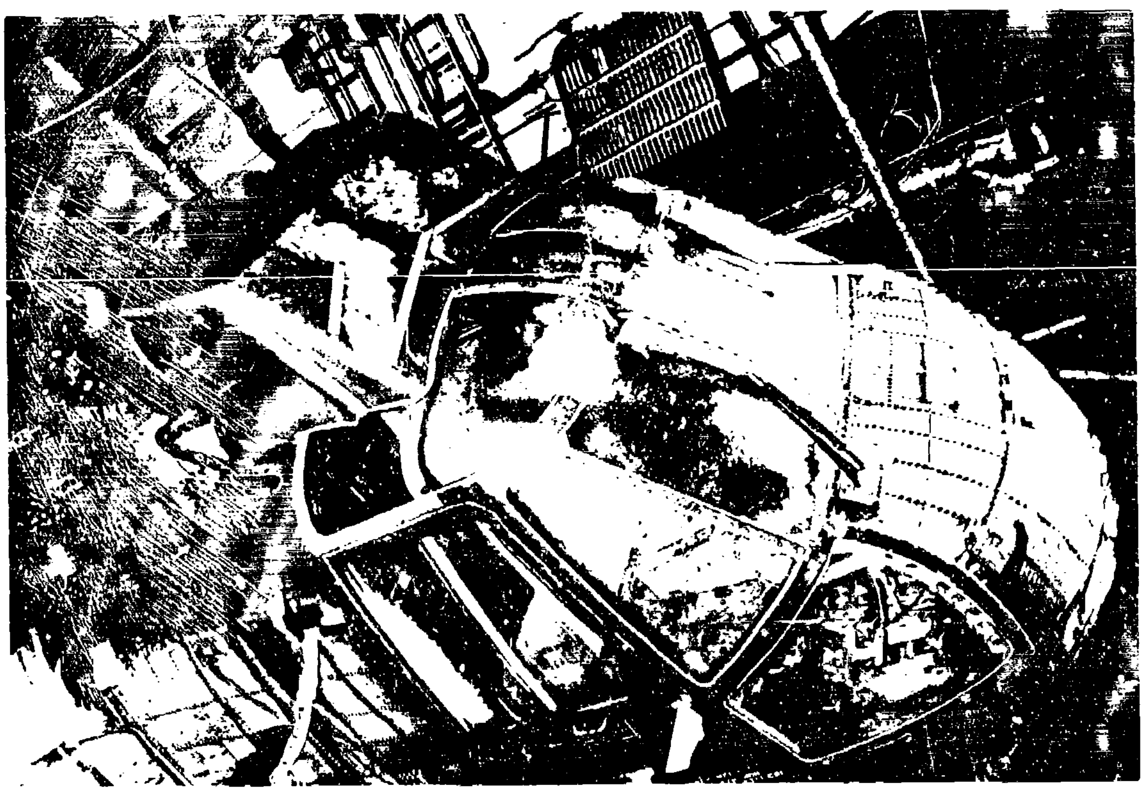

Fig.2 Sea King Forward ice accretion (NGTE test)

Fig.3 Lynx hovering in sand

Fig.4 Lynx hovering in snow

Fig.5 Sea King in high salt corrosion environment (MERC Enterprise Rescue)

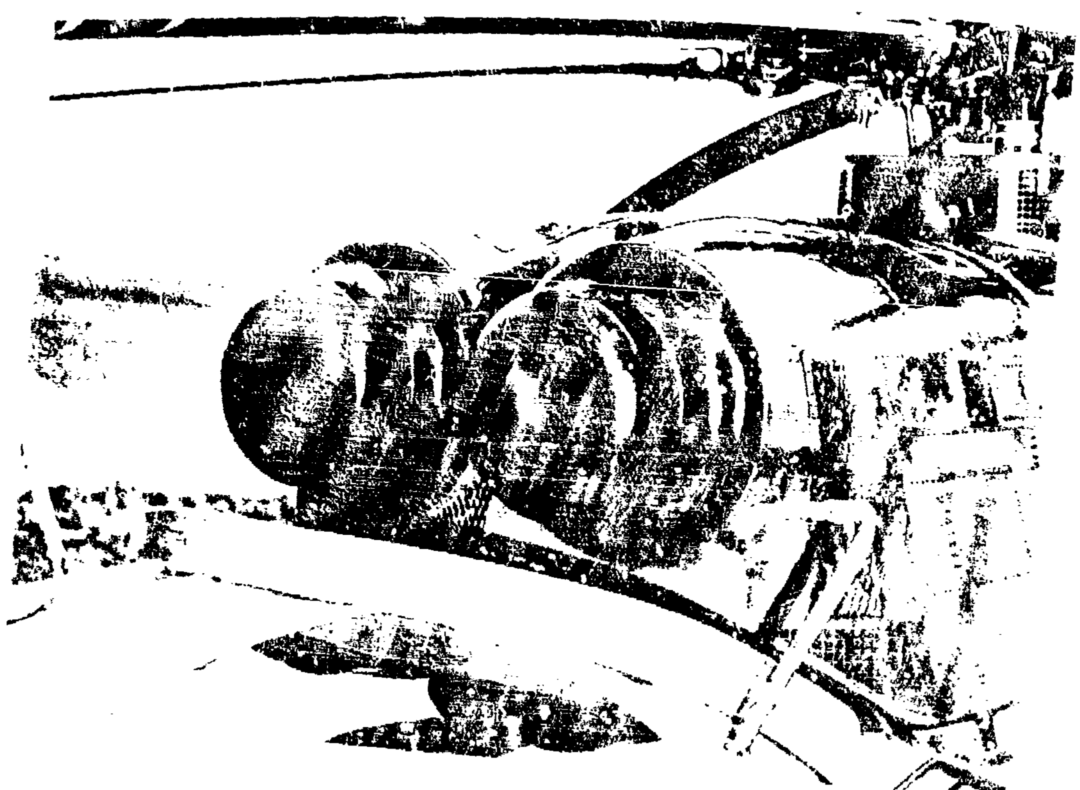

Fig.6 Sea King Mushroom intakes

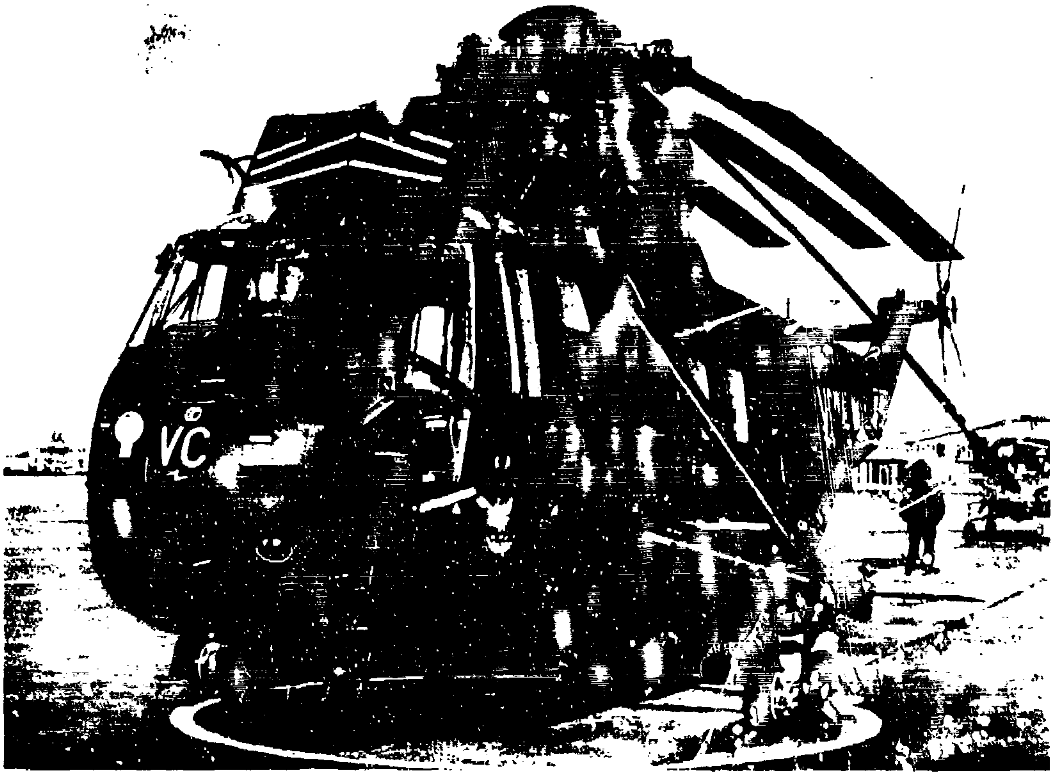

Fig.7 Sea King - FOD intake/TKS fluid strip system

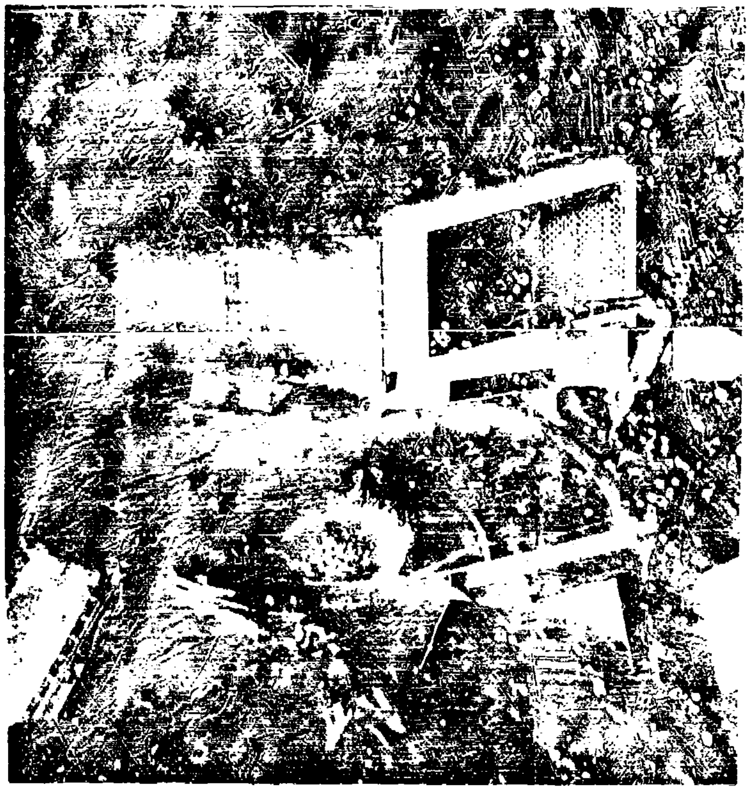

Fig.8 Sea King - Sand filter pack in icing (NGTE test)

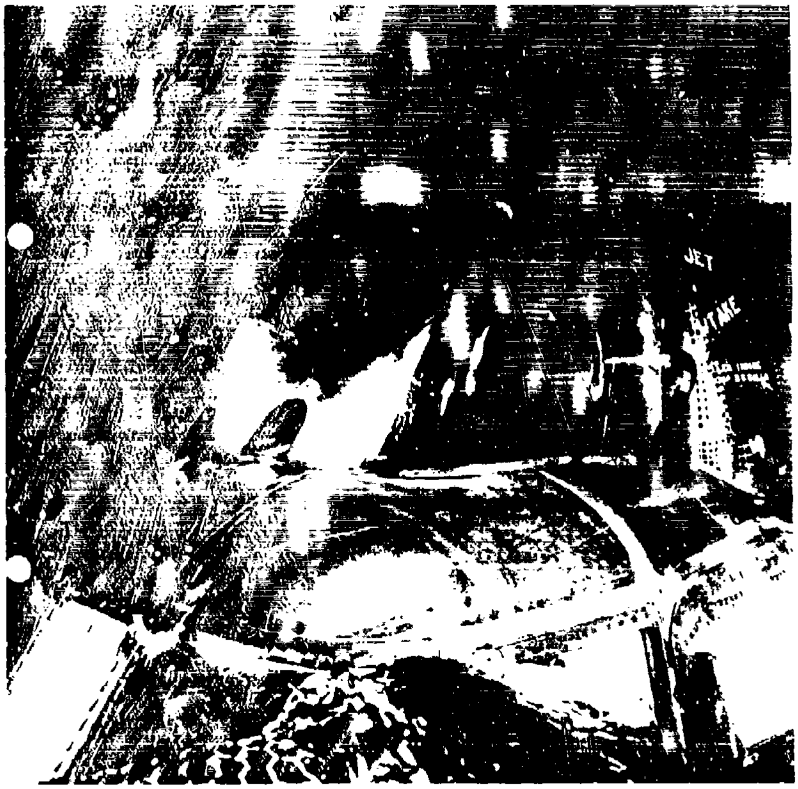

Fig.9 Sea King Heated side intake (NGTE test)

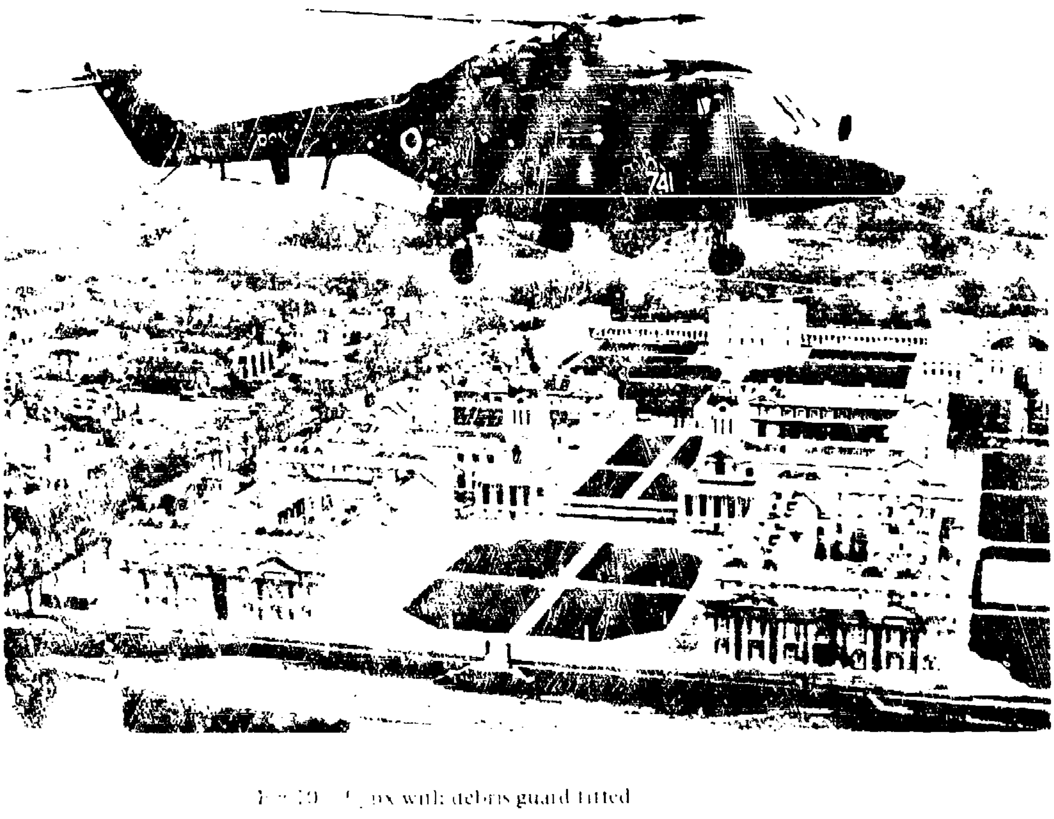

Fig.10 Lynx with debris guard fitted

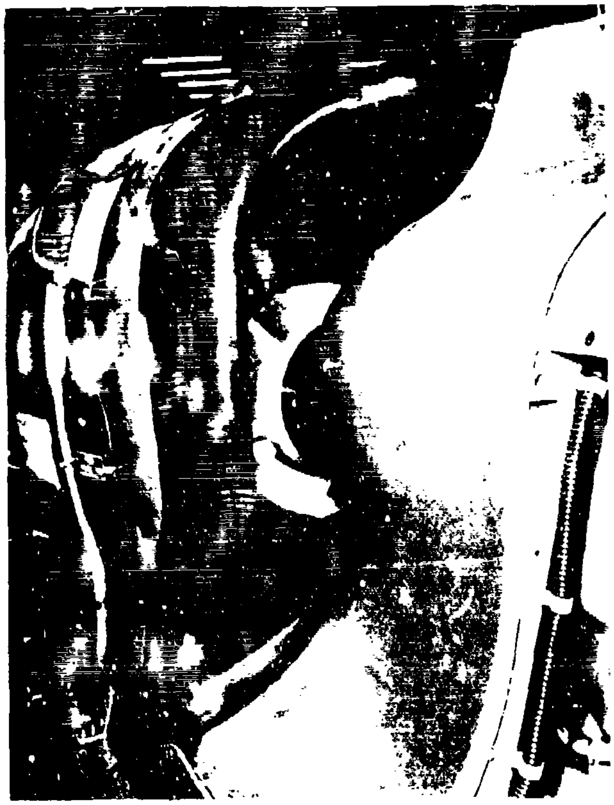

Fig.12 Lynx – Sand filter pack

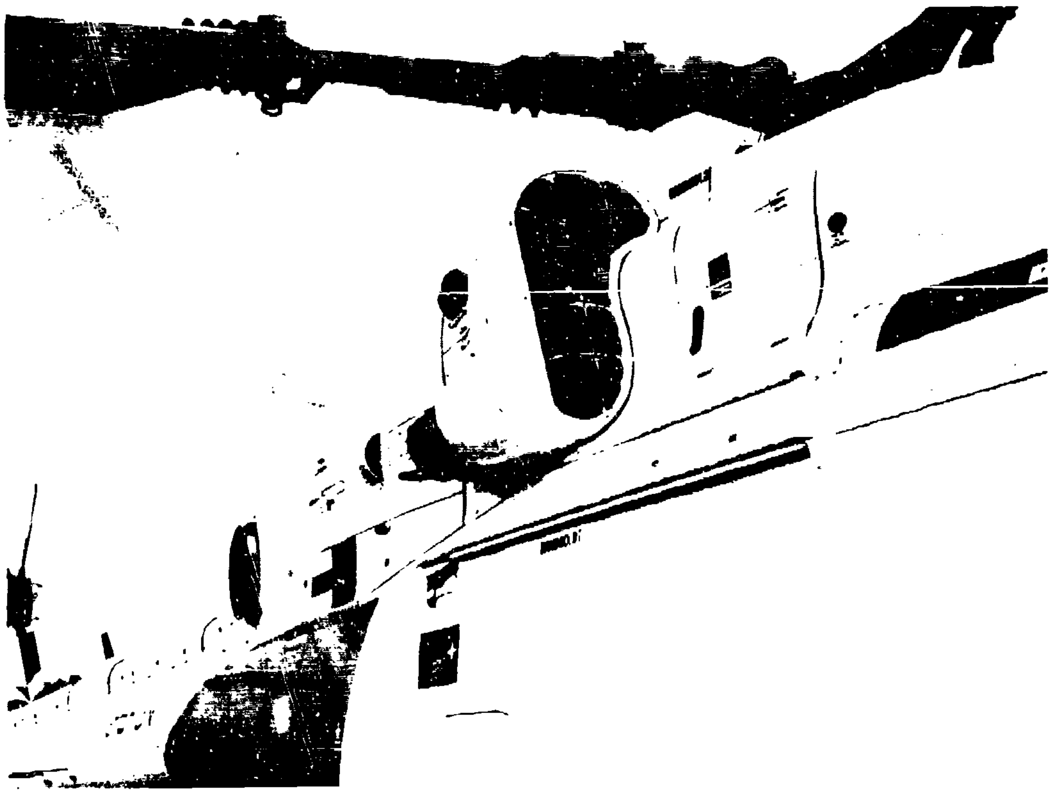

Fig.11 Lynx – Heated air intake

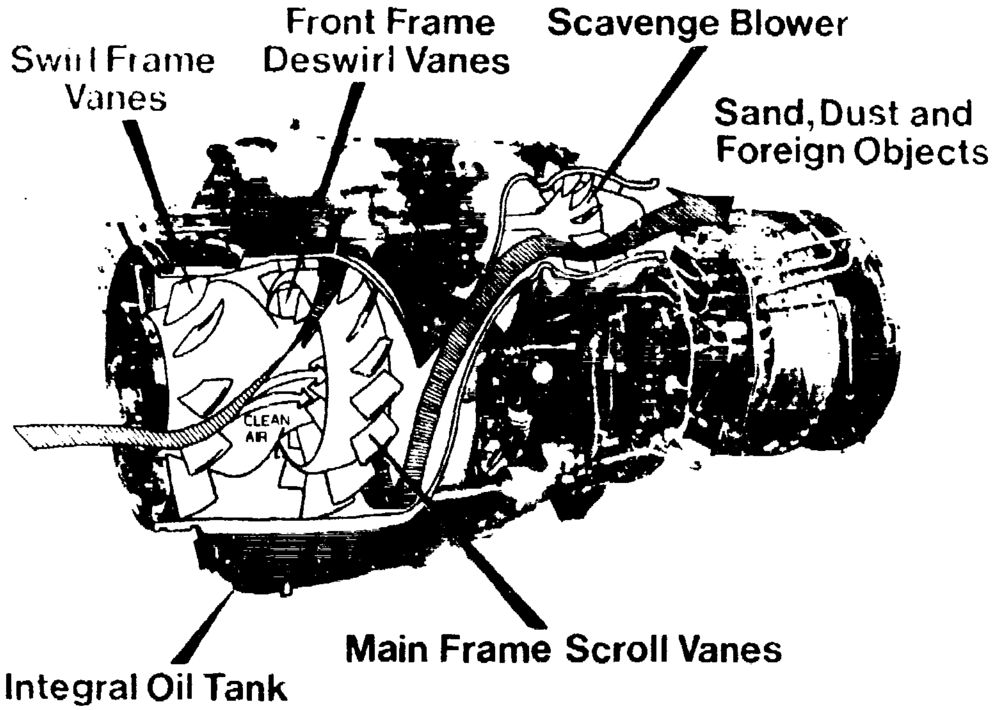

Fig.13 General Electric T700 inlet particle separator

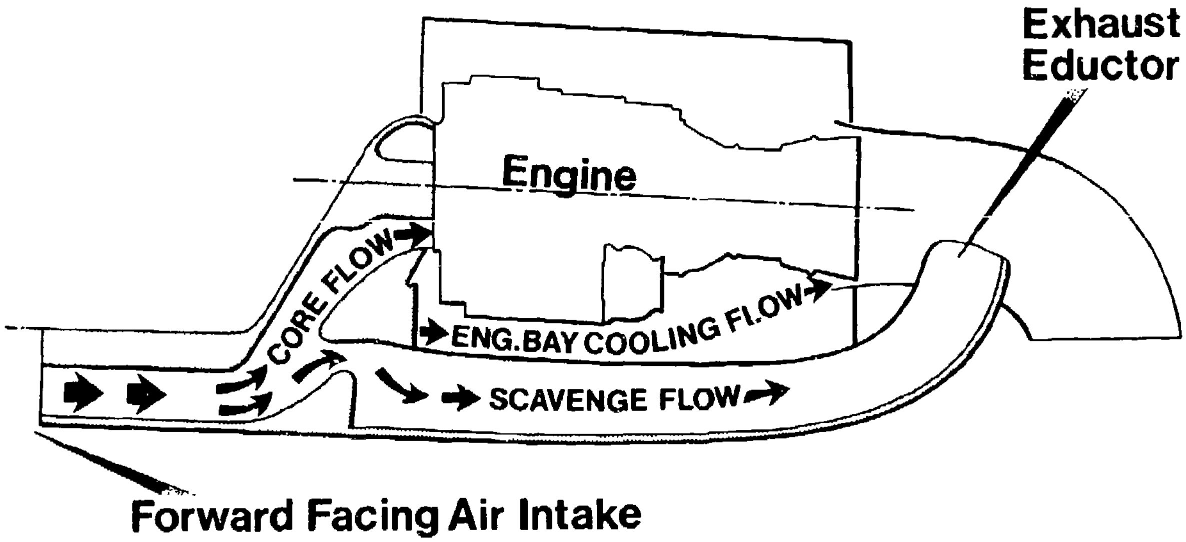

Fig.14 An asymmetric separator proposal

DISCUSSION

K.Rosen, US

Why did you elect to go to the side entry with the T700 which already has a built-in inertial separator?

Author's Reply

For this particular aircraft requiring long operating periods in icing conditions, it is viewed that this design is most appropriate.

REGIME DE DETRESSE SUR HELICOPTERE BIMOTEUR

par

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RESUME - Régime de détresse sur hélicoptère bimoteur.

La tendance à la bimotorisation des hélicoptères civils et militaires, pose la question du choix du niveau de puissance de chaque moteur permettant d'obtenir des performances acceptables en cas de panne d'un moteur, particulièrement aux très faibles vitesses.
L'utilisation d'un régime de détresse, permettant d'obtenir, une fois dans la vie du moteur, un niveau de puissance très élevé pour une courte durée semble être une approche interessante et prometteuse.
Dans une première partie, les relations entre les niveaux de puissance de détresse et les paramètres concernant l'utilisation de l'hélicoptère, masse décollable, plafond, consommation, coût en opération, sont examinés dans le cas d'un projet d'hélicoptère bimoteur de tonnage moyen.
Dans une deuxième partie les perspectives d'homologation, voire de certification civile d'un tel régime sur des moteurs de technologie actuelle sont examinées.

NOTATIONS

PMC : Puissance maximale continue
PMD : Puissance maximale au décollage (5 mn)
PIU : Puissance intermédiaire d'urgence (fonctionnement monomoteur : 30mn)
PMU : Puissance maximale d'urgence (fonctionnement monomoteur : 2 mn 1/2)
PSD : Puissance de "Super Détresse"
T3 : Température d'entrée turbine
T.O.C : Coût total en opération
D.O.C : Coût Direct en opération
Cp : Charge payante
C_K : Prix au kg par km transporté
γ : Coefficient de motorisation : 0m 15°C

$$\gamma = \frac{\text{Puissance thermique des moteurs au décollage}}{\text{Puissance nécessaire au stationnaire H.E.S}}$$

D.E.S : Dans l'effet de sol
H.E.S : Hors effet de sol
Pm : Prix du moteur
T.B.O : Potentiel du moteur (Time Between Overhaul)
Pc : Prix du carburant
C_H : Consommation horaire du moteur.
t : Temps de la mission

1 - INTRODUCTION.

Le choix du niveau de "motorisation" d'un hélicoptère est toujours difficile. Il résulte en effet d'un compromis entre le désir d'obtenir des performances brillantes (par temps chaud ou en altitude) pour pouvoir toucher le maximum de marchés, et les impératifs de rentabilité qui préoccupent tous les utilisateurs en utilisation normale (D.O.C).

Par ailleurs la tendance générale est à la bimotorisation, tant pour les utilisations civiles que militaires. Ceci ne fait qu'accroître la difficulté du compromis, car il est nécessaire d'offrir des performances minimales sur un moteur : ce sont les seules réellement utilisables par les utilisateurs civils catégorie A, et elles déterminent les possibilités opérationnelles de l'hélicoptère militaire (vol tactique à très basse altitude, attente en stationnaire derrière des obstacles naturels...).
Par expérience sur des hélicoptères existant en version mono et bimoteur, nous pouvons considérer que le "prix" de la bimotorisation avec les concepts utilisés aujourd'hui est d'environ + 20 à 25 % sur la consommation kilométrique.

Pour maintenir des caractéristiques opérationnelles comparables (charge utile, range) cette augmentation de consommation s'ajoutant à l'augmentation de masse à vide, nécessite une augmentation de la masse maximale autorisée au décollage ce qui peut entrainer des modifications importantes sur la conception et sur le prix de l'hélicoptère considéré.
Ces écarts sont en partie dus au fait que les moteurs sont dimensionnés pour le régime maximum d'urgence 2,5 mn (car le motoriste doit démontrer 2h 05 mn pendant l'essai d'homologation de 150 h), ce qui entraine des moteurs plus lourds, plus puissants au régime de décollage et ayant une consommation spécifique plus élevée à charge partielle sur 2 moteurs.

En fait, le but recherché est de se protéger au mieux d'un arrêt accidentel d'un moteur pendant certaines phases critiques de vol :

- En utilisation civile le taux d'arrêt moteur en vol se situe autour de 10^{-5} par heure de vol. Mais la probabilité d'arrêt pendant la phase critique de décollage est encore bien plus faible ($< 10^{-7}$): il n'y a aucun cas de ce type répertorié avec les hélicoptères bimoteurs de l'Aerospatiale.

- En utilisation militaire cette probabilité est bien sûr supérieure (pourcentage plus grand de phases critiques de vol, risques d'impact de projectiles...). Cependant, même pour ce type d'utilisation, la probabilité reste telle qu'il est envisageable d'effectuer une inspection ou une révision du moteur ayant dû fonctionner au régime d'urgence à la suite d'une panne de l'autre moteur.

Les questions que nous nous posons et auxquelles cet exposé tente de répondre, sont donc les suivantes : "Ne paie-t-on pas trop cher cette protection contre la panne d'une moteur, compte tenu de sa faible probabilité ? Ne peut-on pas améliorer la situation en allégeant les règles d'homologation des régimes d'urgence, pour pouvoir aller beaucoup plus près des limites thermiques et mécaniques du moteur en cas de panne de l'autre moteur ?".

2 - SITUATION ACTUELLE.

Pour qualifier le niveau de motorisation d'un hélicoptère nous utilisons souvent le "coefficient de motorisation" $\gamma = \frac{\text{Puissance thermique du ou des moteurs - sol - standard}}{\text{Puissance nécessaire stationnaire hors effet de sol}}$

Masse maxi - Sol - Standard.

Ce coefficient varie habituellement entre 1,2 et 1,5 selon le degré de polyvalence recherché pour les hélicoptères concernés.
La fig. N° 1 donne quelques caractéristiques des hélicoptères Aerospatiale de la nouvelle génération.

Le DAUPHIN, avec γ =1,26, est bien adapté aux missions en climat tempéré et à l'off shore, le TWINSTAR (γ = 1,46) et le SUPER PUMA (1,52) sont très bien motorisés et conservent de très bonnes performances dans des conditions climatiques sévères :
Masse maximale décollable HES au standard + 20°C à des altitudes supérieures à 1500 m.

Malgré ces niveaux élevés de motorisation, les performances catégorie A (panne moteur au décollage) restent modestes, surtout pour les procédures "héliport ponctuel".

Pour des versions militaires de ces hélicoptères, le besoin de surmotorisation serait encore accru par :

- l'augmentation des pertes d'installation (protection infra rouge, séparateur de particules)
- les exigences de vol très près du sol pour lequel il est souhaitable de supprimer la "zone de mort" de manière à sauver l'équipage voire l'hélicoptère en cas de panne moteur, quelle que soit l'altitude ou la vitesse d'avancement, y compris le stationnaire.

La situation décrite ci-dessus, qui n'est pas spécifique aux hélicoptères de l'Aerospatiale, est celle que l'on peut obtenir avec des moteurs dont le régime Maxi d'urgence (2,5 mn) est de 0 et 10 % supérieur au régime de décollage.

L'idée d'un régime de détresse, qui, tout en maintenant au niveau actuel les puissances maxi continue et décollage, et la consommation spécifique à régime partiel, permettrait d'obtenir un niveau de puissance nettement supérieur au régime 2,5 mn actuel, 1 fois dans la vie du moteur, n'est pas nouvelle. Elle n'a cependant pas encore débouché sur des applications pratiques en particulier pour les utilisations civiles, pour lesquelles une évolution des réglements s'impose. En effet le problème de la démonstration et la vérification d'une telle "capabilité" de surpuissance reste posé.

Nous pensons que, compte tenu du niveau de fiabilité des moteurs utilisés à l'heure actuelle sur nos hélicoptères, et des impératifs économiques de plus en plus pressants auxquels sont confrontés les utilisateurs, cette notion de régime de détresse est plus que jamais d'actualité et doit être prise sérieusement en considération, aussi bien par les motoristes que par les organismes officiels de certification.

Pour mettre en évidence les avantages d'une telle approche, nous avons, avec la Société TURBOMECA, réalisé une étude sur le thème d'un hélicoptère bimoteur de référence d'une masse maximale de 4 tonnes, équipé d'un nouveau moteur"Y"de conception moderne, qui donne à cet hélicoptère un bon niveau de motorisation selon les critères actuels (fig. 2).

Après avoir déterminé les limites en cas de panne d'un moteur, et déterminé les niveaux de puissance d'urgence souhaitables dans ces conditions, nous avons mis en évidence et tenté de chiffrer l'intérêt de pouvoir obtenir de telles puissances à partir du moteur de base Y plutôt que de définir un nouveau moteur plus puissant qui délivrerait la même puissance au régime 2,5 mn selon les critères actuels d'homologation.

3 - INTERET DU REGIME DE DETRESSE.

Nous avons supposé dans un premier temps que le moteur Y, tel que défini précédemment avec sa puissance de décollage, sa Cs et sa masse était capable de délivrer pendant un temps de 1 minute au moins une "puissance de détresse" pouvant aller nettement au delà de la PMU actuelle.

Nous avons ainsi étudié l'impact du niveau de régime de détresse sur les possibilités opérationnelles de l'hélicoptère .

3.1 - Effet sur les performances.

A l'aide d'un programme de calcul permettant d'étudier le comportement transitoire de l'hélicoptère après la panne d'un moteur (réf. 1), nous avons pour différentes conditions extérieures, différentes masses de l'hélicoptère , déterminé en fonction de la puissance d'urgence disponible les hauteurs H pour lesquelles :

- la poursuite du vol est possible
- le posé vertical amène à une vitesse d'impact* de 6 m/s (train déformé mais appareil non endommagé)

  *de 2,5 m/s (train non déformé).

Les planches (N° 3 à 6) présentent ces résultats pour lesquels la puissance est exprimée en % de la puissance nécessaire au stationnaire hors effet de sol dans les mêmes conditions. Les points intéressants se situent à l'intersection des courbes de poursuite de vol et des courbes de posé(1). Les résultats essentiels ont été portés sur la fig. N° 7 qui récapitule les performances possibles en fonction du niveau de puissance d'urgence, variant de + 10 % à + 60 % par rapport au régime de décollage actuel.

Notons que dans cette étude de simulation le besoin de puissance d'urgence n'a jamais excédé 30 s et que le temps d'accélération du moteur entre le régime de décollage et le régime d'urgence a été fixé à 1 seconde. En dessous de cette valeur les résultats ne sont pas modifiés. Si le temps d'accélération est supérieur à 2 secondes, ce qui pourrait arriver pour de très forts niveaux de surpuissance, les performances obtenues seraient légèrement affectées.

3.2 - Interêt économique du régime de détresse.

La planche n° 2 montre que le moteur Y, avec son régime d'urgence 8,8 % supérieur au décollage n'autorise pratiquement aucune performance sur un moteur, à la masse maxi de 4000 kg pour des conditions altitude, température plus sévères que le standard.

Nous avons donc imaginé qu'il était possible d'autoriser sur ce moteur un régime de "Super détresse" utilisable 1 minute une fois pendant la vie du moteur, dont le niveau est nettement plus élevé que celui du régime maxi d'urgence actuellement défini.

Moteur Y1 dont la PSD est 24 % supérieure au décollage (Δ T3 = 120°C)

Moteur Y2 dont la PSD est 40 % supérieure au décollage (Δ T3 = 200°C)

Pour pouvoir mettre en évidence l'interêt économique d'une telle approche, nous avons comparé ces moteurs Y1 et Y2 à deux moteurs hypothétiques Z_1 et Z_2, de même conception que le moteur Y de référence dont les régimes maxi d'urgence 2,5mn certifiés sont au même niveau que les régimes de détresse des moteurs Y1 et Y2 (voir pl. 8).

Ces moteurs Z_1 et Z_2 sont des moteurs "homothétiques" au moteur de référence Y et délivrent leurs puissances PMD et PMU aux mêmes températures T3 que le moteur Y, leur consommation à régime partiel est légèrement améliorée à 60 % de la PMD par l'effet d'échelle, mais dégradée à puissance donnée (400 kW).

Nous avons calculé, pour ces différents moteurs :

- le coût total en opération, qui tient compte de la consommation et du prix des moteurs

  $$T.O.C = A + B\ Pm + C\ \frac{Pm}{TBO} + Pc.CH$$

- la charge payante qui tient compte :
  - du niveau de puissance par le biais de la masse décollable, jusqu'à concurrence de la masse maximale
  - de la consommation
  - de la masse de l'installation motrice (masse moteur plus répercussions sur hélicoptère)

  $$Cp = \underbrace{M_T}_{\text{Masse maxi}} - \underbrace{M_V}_{\text{Masse à vide sans moteurs}} - \underbrace{m1}_{\text{masses moteurs}} - CH.t$$

- le coût du kilogramme transporté par kilomètre qui donne une idée de la rentabilité globale de l'hélicoptère

  $$C_K = \frac{T.O.C}{Cp.V}$$

Les résultats sont donnés sur les fig. N° 10 à 12.

Avec les procédures catégorie A héliport ponctuel, pour lesquelles le régime limite actuel est la PMU, les gains obtenus sont assez spectaculaires sur le coût du Kg/Km.

Pour les moteurs 1 (PSD ou PMU =1,24 PMD) on obtient déjà une réduction du coût (Kg/Km) variant de 50 % (Zp = 0 to = 15°) à 70 % (Zp = 0 to = 35°C).

Par ailleurs les moteurs Y, avec régime "Super Détresse" offrent un avantage pouvant atteindre 25 % par rapport aux moteurs Z n'ayant qu'un régime d'urgence.

(1) Ils correspondent en fait à la suppression de la zone d'insécurité.

Dans le cas des procédures héliport dégagé, pour lesquelles le régime limite est la PIU (taux de montée imposé), l'appareil pouvant, dans un premier temps, prendre de la vitesse horizontale pour réduire la puissance nécessaire, les résultats sont sensiblement différents.

Les moteurs Z ont une capabilité de PIU accrue. Nous avons supposé que leur PIU était égale à la PMD. Les performances et la rentabilité de l'hélicoptère sont donc améliorées.
Pour les moteurs Y les performances et la rentabilité de l'hélicoptère ne sont améliorées que si le régime PIU est relevé, simultanément au régime PSD.

Dans l'hypothèse ou la PIU est relevée au niveau des moteurs Z on retrouve des gains comparables à ceux sur héliport ponctuel entre moteurs Y et moteurs Z.

En fait, pour les procédures civiles catégorie A, ces résultats ne doivent être considérés que pour leur ordre de grandeur, une étude particulière à chaque hélicoptère devant être faite pour savoir jusqu'à quel niveau il est interessant d'augmenter la PSD sans augmenter la PIU.

Par contre pour des applications "militaires" où il n'y a pas de poursuite du vol avec taux de montée imposé à la PIU, seul le niveau de PSD est déterminant pour la survie de l'hélicoptère et les résultats présentés (charge utile /Prix Kg/Km) sont directement utilisables.

4 - POSSIBILITES REELLES DES MOTEURS.

Le problème principal lié au régime de détresse est la résistance des pales de turbine au phénomène de fluage (Température et vitesse).

Nous pouvons considérer que les températures T3 actuellement retenues pour le décollage correspondent à une durée de vie calculée de 1000 h dans le cas du moteur Y de référence non refroidi, et ayant été dimensionné pour atteindre des TBO élevés.

La courbe n° 13 montre l'évolution de cette durée de vie calculée avec le niveau de température, et la surpuissance que l'on peut attendre.

A 22 % de surpuissance la durée de vie calculée tombe à 10 h, à 33 % à 1 h et à 50 % elle n'est plus que de quelques minutes.

A l'aide de ces résultats de calcul et de l'expérience acquise par TURBOMECA au cours de nombreuses homologations antérieures, et de 25 millions d'heures de vol, nous pouvons dire que, à condition de réduire la sévérité de l'essai d'homologation, les moteurs de conception actuelle ont une capabilité de régime de détresse (1 fois dans la vie du moteur) supérieure à 120 % du régime décollage :

- entre 20 % et 25 % de surpuissance (ce qui correspond environ à 10 h de durée de vie calculée) avec la possibilité d'utiliser le moteur au régime intermédiaire d'urgence après l'utilisation de la détresse (cas de poursuite du vol)

- entre 25 % et 30 % de surpuissance ou plus (environ 1 heure de durée de vie calculée) avec arrêt du moteur après l'utilisation du régime de détresse.

Au delà de 25 % la prévision par calcul devient difficile. Cependant il n'est pas exclu que des essais sur moteur réel puissent démontrer des capabilités de régime de détresse supérieures.

Il est par ailleurs intéressant de souligner que pour la même élévation de température T3, l'augmentation relative de puissance est plus importante par temps chaud qu'en standard.

Par exemple pour le moteur Y de référence nous obtenons, les chiffres suivants :

t_0	Régime	Puissance	t_3	Vitesse générateur
15°	PMD	100 %	t_3 Réf.	100 %
	PSD	120 %	t_3 Réf. + 90°C	104,5 %
35°	PMD	100 %	t_3 Réf.	100,5 %
	PSD	125 %	t_3 Réf. + 90°C	103,6 %

Donc en fait, une surpuissance de 20 % en conditions standard représente 25 à 30 % de surpuissance par temps chaud, dans des conditions où justement le besoin est le plus important.

Dans le cas particulier du moteur Y nous pouvons conclure qu'une surtempérature de 110°C par rapport au décollage est possible (10 h de durée de vie calculée) ce qui permet une surpuissance de 30 % à to = 35°C.

La définition d'un tel régime pourrait être :

Puissance de Super détresse :

Puissance maximale approuvée pour l'utilisation en cas de panne d'un moteur au cours d'un décollage ou d'un atterrissage et limité à une seule période d'utilisation ininterrompue d'une durée maximale d'une minute.

Cette utilisation est accompagnée de l'obligation de déposer le moteur pour inspection complète ou révision générale.

Compte tenu des niveaux de température atteints au cours de l'utilisation d'un tel régime, il est nécessaire de s'entourer d'un certain nombre de précautions de manière à éviter les utilisations intempestives ou abusives :

- armement à la demande du pilote (la limite normale restant la PMU actuelle)
- interdiction de redémarrer le moteur après utilisation de ce régime, ou utilisation d'un calculateur d'endommagement.

Le test, permettant de vérifier que le moteur est bien capable de cette puissance, devrait se limiter à :

- un contrôle de l'état du moteur à des régimes inférieurs
- une vérification rapide du régime générateur atteint. L'accélération du moteur du régime de décollage ou régime de super détresse se faisant en moins d'une seconde, ce test devrait pouvoir se faire périodiquement sans entamer de manière significative le potentiel de vie du moteur. Il est certain que, là encore, la surveillance du moteur par calculateur d'endommagement apporterait une sécurité supplémentaire.

Quant à l'homologation d'un tel régime, nous pensons qu'elle pourrait se dérouler comme suit :

Première partie :

Exécution de l'essai d'endurance de 150 heures tel que défini par la réglementation actuelle, avec, en plus, par phase de 6 heures, deux affichages durant 10 secondes de la vitesse de rotation correspondant au régime de Super détresse.

Deuxième partie :

- Essai continu de 5 fois le cycle ci-après :
  - 5 mn. Ralenti sol
  - 1 mn. Décollage
  - 1 mn. Super détresse
  - 30 mn. Intermédiaire d'urgence
  - 60 mn. Régime intermédiaire (croisière).
- Démontage et examen final :

Essais complémentaires de type.

Pour valider complètement un moteur prévu avec le régime de Super détresse un certain nombre des essais particuliers de type doivent être adaptés aux conditions nouvelles. Ce sont :

(A) Essais de fonctionnement (en particulier pompage).

(B) Essais de vibrations :

(C) Contrôle préliminaire et final de la puissance au régime de Super détresse.

Il est convenu de ne pas faire de mesure de puissance, au régime de Super détresse, lors des contrôles préliminaires et finals des caractéristiques de l'essai d'endurance de type de 150 heures.

NOTA : Ceci implique qu'un essai particulier, approprié, soit fait sur un autre moteur pour démontrer la puissance déclarée pour le régime de Super détresse.

(D) Essai d'intégrité en survitesse :

Essai de 2 mn à la température devant turbine déclarée pour le régime de Super détresse et à une vitesse de rotation supérieure de 10 % à celle du régime de Super détresse.

Si l'on considère la faible probabilité pour que le dispositif de régulation et de commande tombe en panne précisément au moment de l'affichage de la vitesse de Super détresse et si l'on tient compte du fait qu'après l'emploi de ce régime, le moteur devra obligatoirement être déposé et examiné, il semble raisonnable d'adopter la valeur de 10 % de survitesse par rapport à la vitesse de Super détresse.

(E) Essai d'intégrité en surtempérature :

Essai de 5 mn. à la vitesse de rotation du régime Super détresse et à une température devant turbine dépassant d'au-moins 20°C la température maximale déclarée pour le régime.

CONCLUSION.

L'utilisation d'un régime de "Super Détresse", utilisable une fois pendant une minute, permet d'augmenter très significativement les performances sur un moteur d'un hélicoptère bimoteur. L'amélioration de rentabilité qui en découle est très attractive.

Les moteurs de conception actuelle ont une capabilité importante de surpuissance atteignant 30 % par temps chaud, l'homologation et l'utilisation de tels régimes ne posant pas de problèmes insurmontables.

Il est donc nécessaire d'entamer le processus d'évolution des réglements d'homologation des moteurs pour permettre l'utilisation de tels régimes.

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NOUVELLE GENERATION DE BIMOTEURS AEROSPATIALE
PERFORMAI CES COMPAREES

		AS 355 F TWINSTAR	AS 365 N DAUPHIN	AS 332 SUPER PUMA
MASSE MAXIMALE		2300 Kg	3850 Kg	7800 Kg
MOTEUR INSTALLE		ALLISON C20F	TURBOMECA ARRIEL 1C	TURBOMECA MAKILA 1A
PUISSANCES (kW)	PMD	313	482	1240
	PIU	313	612	1240
	PMU	313	622	1310
COEFFICIENT DE MOTORISATION (PUISSANCE INSTALLEE/ PUISSANCE NECESSAIRE) STANDARD SOL		1.46	1.20	1.82
PLAFOND HES	Std	2400 m	1000 m	2400 m
	Std + 20	1560 m	0 m	1700 m
PERFORMANCES	TO MAXI A Zp 0 HELIPORT DEGAGE	35°C	37°C	>50°C
CATEGORIE A	HELIPORT PONCTUEL	2100 Kg à Zp 0m 15 °C	3340 Kg à Zp 0 m 15 °C	7800 Kg à 0 m 24 °C 8000 Kg à 0 m 15 °C

FIGURE 1

PERFORMANDES DE L'HELICOPTERE DE REFERENCE

MASSE MAXIMALE		4000 Kg
PUISSANCE D'UN MOTEUR Y	PMD	625 kW
	PIU	625 kW
	PMU	680 kW
COEFFICIENT DE MOTORISATION		1.45
PLAFOND HES	Std	2 300 m
	Std + 20	900 m
CATEGORIE A	TO MAXI Zp = 0 m HELIPORT DEGAGE	43°C
	TO MAXI Zp = 0 m HELIPORT PONCTUEL	– 4°C 3800 Kg à 15° C

FIGURE 2

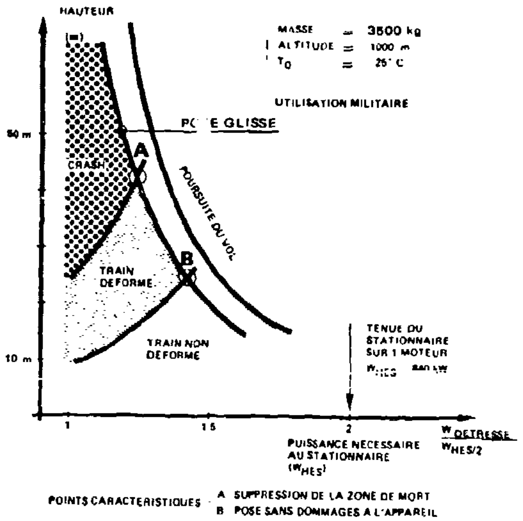


FIGURE 3

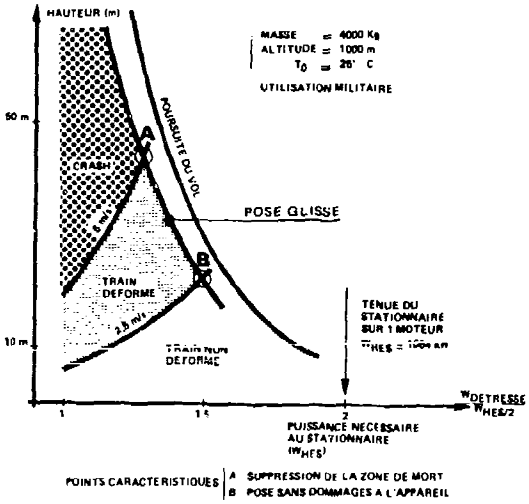


FIGURE 4

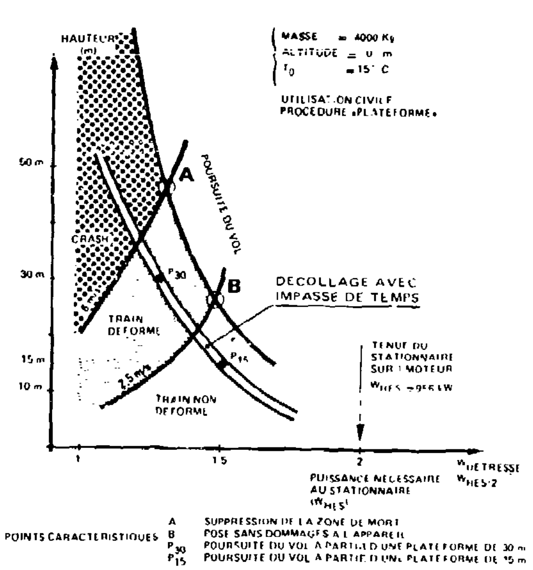


FIGURE 5

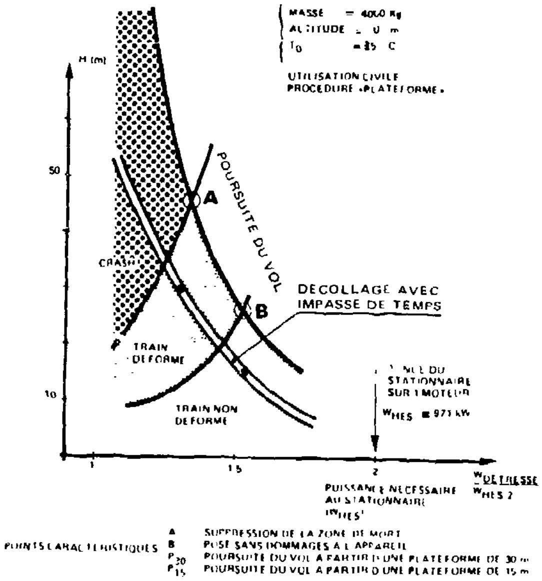


FIGURE 6

IMPACT DU REGIME D'URGENCE SUR LES PERFORMANCES DE L'HELICOPTERE DE REFERENCE

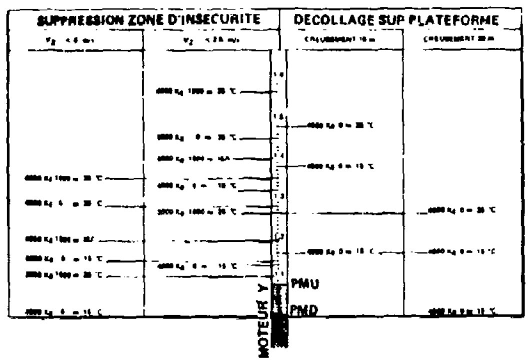
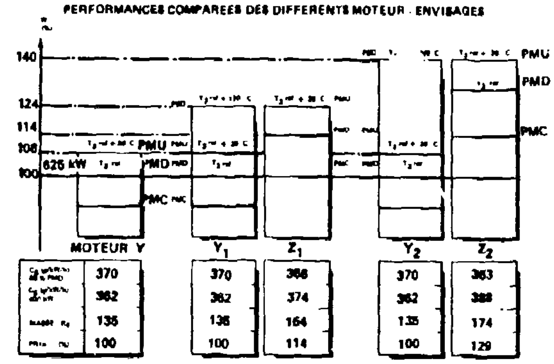


FIGURE 7

PERFORMANCES COMPAREES DES DIFFERENTS MOTEUR ENVISAGES



PSD : PUISSANCE «SUPER DETRESSE»

FIGURE 8

CHARGE PAYANTE ET COUT D'EXPLOITATION

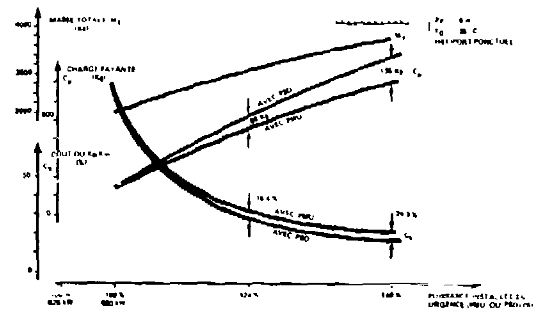


FIGURE 10

DEFINITIONS

COUT TOTAL EN OPERATION

$$T.O.C. = A + B.\ Pm + C.\frac{Pm}{TBO} + Pc.C_M$$

- A : COUT DE L'HELICOPTERE, FRAIS FINANCIERS, MAINTENANCE, EQUIPAGE (HORS MOTEUR)
- Pm : COUT DU MOTEUR
- TBO : POTENTIEL DU MOTEUR
- Pc : PRIX DU CARBURANT
- C_M : CONSOMMATION HORAIRE DU MOTEUR

CHARGE PAYANTE TRANSPORTABLE

$$Cp = M_T - M_V - 2\, m_1 - C_M . t$$

COUT DU KG PAR KM TRANSPORTE

$$C_K = \frac{T.O.C.}{Cp . V_{..}}$$

FIGURE 9

CHARGE PAYANTE ET COUT D'UTILISATION

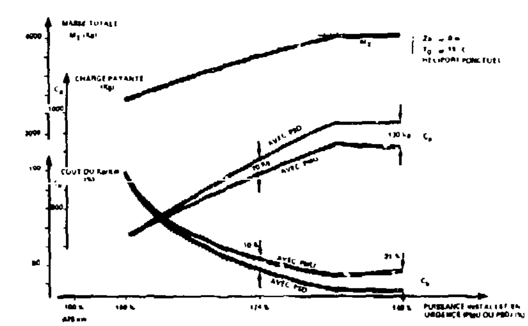


FIGURE 12

CHARGE PAYANTE ET COUT D'EXPLOITATION

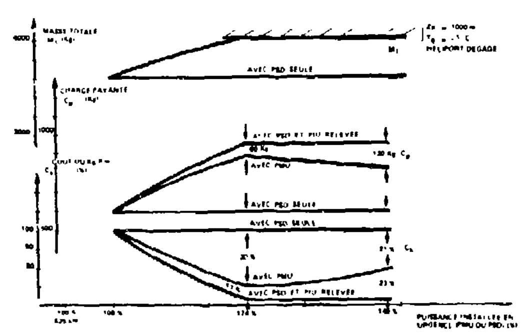


FIGURE 11

DUREE DE VIE CALCULEE DES TURBINES

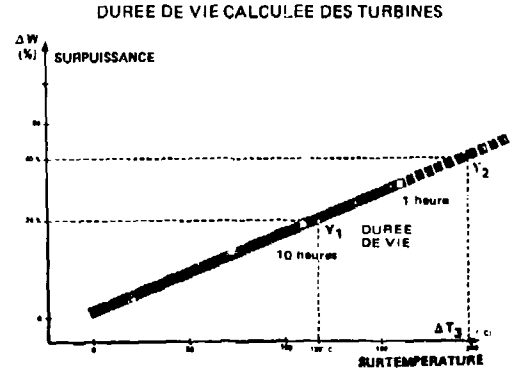


FIGURE 13

DISCUSSION

K.Rosen, US

Would you plan on having an automatic increase on the temperature limiter in the event of an engine failure?

Author's Reply

It is agreed that such a limiter increase must be provided and must be automatic.

R. de Gaye, UK

Does the Makila engine have provision to reset the maximum level of power on the remaining engine in the event of a single-engine failure?

Author's Reply

Not at this time. However, the engine does have capability of an emergency rating higher than the normal maximum contingency rating. Such operation has yet to be demonstrated.

R. de Gaye, UK

Could you comment on the need for such an emergency rating device since gearbox torque limits could also be used as a controlling device?

Author's Reply

In the case where you do not have a torque limiter, control of gearbox torque is certainly suitable and has been used. However again, when an automatic temperature limiter is utilized, provision for by-passing this limiter during an engine failure must be provided.

ENGINE/DRIVE/AIRFRAME COMPATIBILITY — A WAY OF LIFE

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ABSTRACT

Virtually all of the helicopter manufacturers have encountered some form of engine/drive/airframe compatibility situation for which there was concern during the early development stages or during production of a new design. Studies and analyses of such compatibility problems have now become commonplace and can be considered a way of life in the helicopter industry. This paper discusses various compatibility encounters on Boeing Vertol helicopters, their analyses, and their solutions as affected by current technology and engine/airframe manufacturer relationships.

GENERAL

The concept of system compatibility is certainly not a new one. Perhaps one of the earliest problems encountered on helicopters was that of ground resonance. This necessitated the need for evaluating the compatibility of the rotor system, rotor damping, airframe inertia, oleo placement on the aircraft, and oleo damping characteristics. Because of these early experiences ground resonance is now well understood; the analytical tools are available and are properly considered in the early design stages of the helicopter. Since these systems are usually all under the control of the airframe manufacturer, there is little need to interface with other subcontractors.

With the sophistication of current turbine-powered helicopters, however, have come many new problems involving the compatibility of the major multidegree-of-freedom systems: rotor, control, airframe, drive train, and powerplant. It is unfortunate that problems related to dynamic interface items were often among the last to be encountered; they therefore become the most costly to correct.

Extremely close communication is required today, not only between the several technical disciplines but also between two or more manufacturers, if a problem is to be solved before the product reaches the hardware stage.

Fortunately papers have been written and reports issued that summarize the system compatibility experiences of the various helicopter manufacturers. A wealth of information on this subject is available in References 1 through 7.

The remainder of this paper is devoted to a detailed review of various compatibility encounters at the Boeing Vertol Company over the years from 1964 to the present. The solutions to the various problems, for the most part, were not highly sophisticated; in fact, most solutions were attained from the application of basic engineering principles to state-of-the-art analytical and test techniques. Specific examples have been selected that relate to each of the various helicopter systems.

ROTOR/ENGINE COMPATIBILITY

Situation

During a twin-engine rotor start, three lag dampers were damaged due to overloading. This occurred on the CH-46E helicopter in 1975 shortly after higher-powered T58-GE-16 engines were installed.

Investigation

The rotor lag dampers act as stops during rotor low-speed accelerations. The allowable torque which the dampers can withstand is a function of the rotor speed, since centrifugal force on the blades provides a relieving moment about the lag hinge and therefore decreases the amount of torque which must be absorbed by the lag damper. The rotor startup procedure and fuel control management were therefore key factors affecting the magnitude of damper loads.

The engine fuel control lever, which normally has a linear relationship to torque from ground idle through maximum power during steady-state operation, was found to exhibit nonlinear characteristics during transients. These transients resulted from various limiting functions within the engine control system and the required switchover from gas-generator power control to power-turbine speed governing. Increased input on the control shaft increased the engine output and decreased the time to reach operating speed. Because of the acceleration/time effects the pilot had very little control of the torque peak.

Solution

A detent, or gate, between the GROUND IDLE and FLY positions was provided to enable the pilot to accurately dwell at a safe engine power setting and thereby reduce the possibility of an overtorque. Figure 1 shows the relationship between the maximum engine torque developed and the engine control shaft setting, including the effect of ambient temperature. The detent set at the 40-degree position of the engine control lever provided assurance that the torque would remain below the continuous twin-engine transmission torque limit of 100 percent down to 0°F ambient. In addition, it was required that the rotors reach a speed of 80 rpm (30 percent) prior to advancing the condition levers from the GROUND IDLE detent. This insures that the blades develop sufficient centrifugal force to eliminate the possibility of overloading the lag dampers when rotor shaft torque is increased.

Commentary

The CH-46 helicopter must operate from aircraft carriers. For such operations it is desirable to bring the rotor blades up to speed as quickly as possible. This is done with articulated rotors to prevent excessive blade flapping that might cause blade contact with the fuselage, due to the gusty, turbulent air conditions on the carrier.

In this particular case the desire to reduce startup time was inconsistent with the loads generated during startup and therefore a compromise was necessary. The transient analyses performed served only as a guide in understanding trends. The final solution for starting procedures was established, however, through testing and service experience.

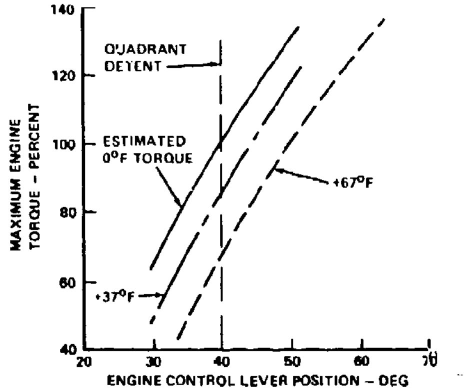

Figure 1. Peak Engine Torque Resulting From Rapid Advance From Ground Idle

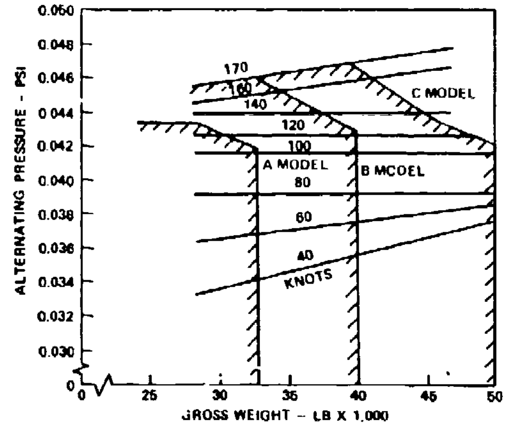

Figure 2. Measured Alternating Air Pressures

AIRFRAME/ROTOR/DRIVE COMPATIBILITY

Situation

A considerable amount of fatigue cracking had occurred in the secondary and redundant structure of the airframes on CH-47A, B, and C helicopters during 1969. Although the cracking was not considered to be a safety-of-flight problem, it did have an effect on maintenance manhours and therefore became the subject of a significant effort to establish the cause and a solution.

Investigation

The interacting systems were found to be the rotor, drive, and airframe. Flight-test data indicated the presence of 3- and 6-per-rev dynamic pressure fluctuations due to rotor blade passage over the fuselage. These fluctuations, as shown in Figure 2, were dependent on both helicopter forward speed and gross weight. Figure 3 shows the crown frame stresses resulting from the pressure fluctuations and their relationship to the allowable stresses.

In addition to airframe cracking, there was also an excessive failure rate of the synchronizing shaft support spring mounts. Analysis indicated that the aerodynamic pulses were exciting the crown frames and the shaft mounting structure in a coupled mode at near-resonant conditions.

Solution

The most simple and straightforward solution in this case was to detune the crown frame structure by stiffening the frames and by providing proper synchronizing shaft support mount stiffening. The effect of the structural modifications on stresses is shown in Figure 4. With stress levels reduced below the fatigue allowables, the problem was eliminated.

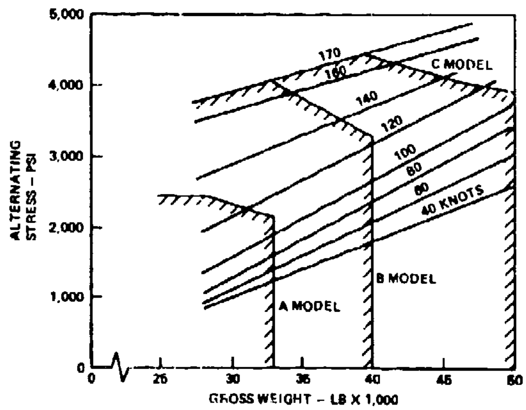

Figure 3. Measured Crown Frame Stresses

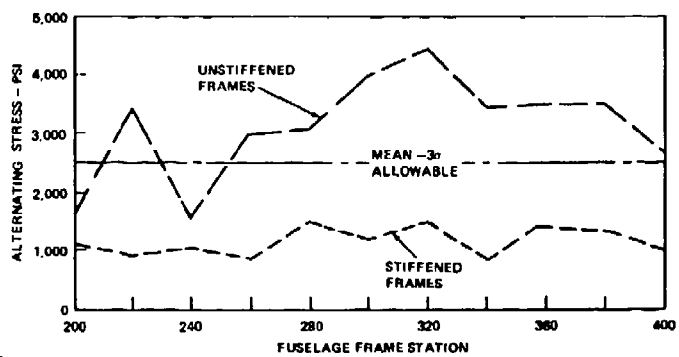

Figure 4. Top-of-Scatter Flight Stress Measurements at V_{NE}

Commentary

At that time the use of finite-element analysis for the evaluation of airframe structure, both from a dynamic analysis and strength analysis viewpoint, was in its infancy. Today, however, such methods are being used by most helicopter manufacturers and can be used to evaluate airframe frequencies, modes, and stresses due to applied pressure fluctuations. Such calculations now can and should be performed during the design stage and not after the aircraft is in production.

DRIVE/AIRFRAME COMPATIBILITY

Situation

In 1964 a failure occurred on the connecting quill shaft that transmits torque from the combiner transmission to the aft transmission on the Boeing Vertol 107 helicopter. No metallurgical deficiencies were found and the original qualification testing did not reveal any potential structural problem with the quill.

Investigation

The connecting quill shaft serves to transmit torque between the combiner transmission and the aft rotor transmission of the V-107/CH-46 helicopter. Physically there are two complete and separate transmissions, but they are connected by a piloted and bolted flange into one unit as shown in Figure 5. Because of the arrangement of mounts (four on the rotor transmission and two on the combiner transmission) and the airframe stiffness, rotor torque is reacted at the combiner mounts as well as at the rotor transmission mounts. The resulting deflections of the airframe structure as well as internal deflections in each transmission impose displacements at each end of the quill shaft. Therefore, in addition to its primary torque loading, the connecting quill shaft is subjected to secondary (and, in fact, far more significant) alternating bending loads which can quite easily exceed its endurance limit if not adequately controlled. This is illustrated in Figure 6.

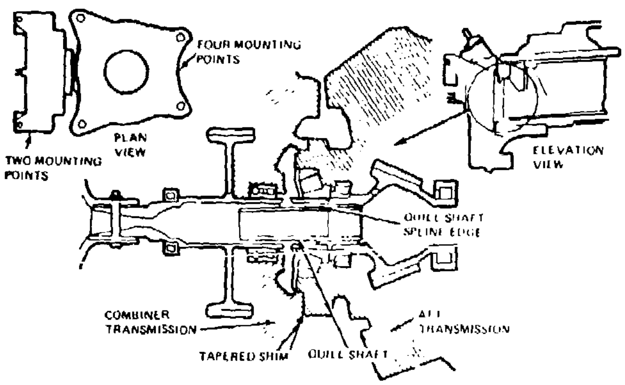

Figure 5. Transmission/Airframe Mounting Configuration

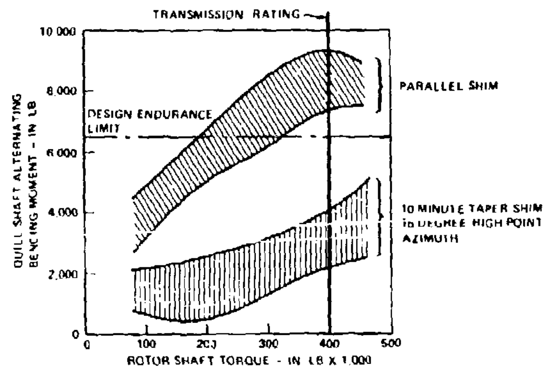

Figure 6. Quill Shaft Alternating Bending Moment Versus Rotor Shaft Torque

Solution

The induced bending was controlled by introducing a tapered shim at the bolted flange between the two transmissions. This shim has the effect of biasing or building in a fixed misalignment that is opposed to the displacements induced by the rotor torque.

In addition to the use of a tapered biasing shim, an installation of this type requires that manufacturing tolerances be held within tight limits. In this case the concentricity and perpendicularity of the piloted flange surfaces on both transmissions are controlled so that they cannot offset the effect of shimming. Careful control of the complete transmission installation in the airframe was necessary to prevent unwanted distortions since the mounting surface consists of six points in the same plane.

Commentary

Once again, the fact that this helicopter was designed before the age of sophisticated computers and finite-element analytical programs did not permit accurate calculations of the structural airframe stiffnesses and deflections during the design stage. This capabilty does exist today and is being used during the design stage to establish realistic load distributions and displacements of transmissions mounted in a flexible airframe structure.

ENGINE/DRIVE/ROTOR COMPATIBILITY

Situation

In 1969 when the model CH-47C helicopters with T55-L-7C engines were converted to the full C model by incorporating T55-L-11 engines, fluctuations in engine and rotor shaft torque were observed. The fluctuations were audible and were disconcerting to the pilot and crew. The primary differences between the -7C and -11 engines were a 50-percent increase in power turbine inertia and a new, more responsive, fuel control configuration.

Investigation

A computer simulation of the CH-47C rotor system with the T55-L-11 turbine engine was made before initial flight testing began. The simulation indicated favorable engine control stability. Initial ground and flight testing with the new installation resulted in unacceptable oscillations in torque and rpm. The oscillations were observed only in hover and on the ground, not in forward flight.

A study of pertinent parameters indicated that the slope of the blade lag damper force-velocity curve below the preload force level had a significant effect on torsional stability. When this curve was artificially stiffened beyond its actual limits, as shown in Figure 7, the analysis reproduced the oscillation. Once the oscillation had been simulated, the analysis showed that it could be eliminated by reducing the steady-state gain and using a slower fuel control response.

Flight tests were then conducted with a set of lag dampers having a significantly reduced preload slope together with the original fuel controls. The tests showed that the oscillation was suppressed. Since the lag damping capacity had been reduced, however, the ground resonance margins were degraded and damper modification was rejected as a solution.

Solution

Flight testing was conducted on a fuel control with a 30-percent reduction in steady-state gain and an increase in time constant. This provided acceptable engine torque stability in the cold and over the entire engine operating envelope. Pilots noted that the engine response to input power demands was not perceptibly degraded with these slowed-down controls. These modifications were then incorporated in the CH-47C fleet and have provided an acceptable production fix.

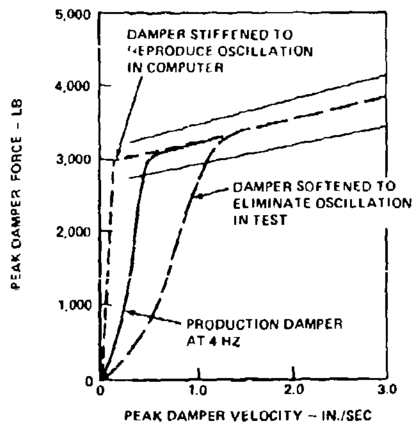

Figure 7. Lag Damper Force-Velocity Curves

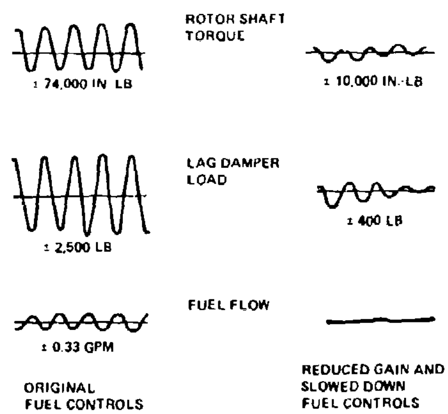

Figure 8. Final Torque Oscillation Simulation

Commentary

The rotor/engine control mathematical model required an accurate model of the lag damper which included the hydraulic spring effect of the damper. Inclusion of this lag damper mathematical model into the torsional stability computer simulation accurately reproduced the torque oscillation with the original fuel controls. Frequency of oscillation, phasing, and magnitude of damper force, shaft torque oscillation, and fuel flow fluctuations were simulated accurately. The final simulation can be seen in Figure 8.

During the program, it became apparent that the engine and airframe manufacturers can easily coordinate their efforts to prevent this type of incompatibility. The airframe manufacturer can provide rotor and drive parameters to the engine manufacturer and the latter can provide a model of the engine and fuel control system.

ACCESSORY/DRIVE COMPATIBILITY

Situation

During the preliminary airworthiness evaluation of the CH-47D helicopter in 1979, a failure of the forward transmission oil cooler fan impeller occurred. The fan impeller was replaced with a new unit but within a few additional flight hours this fan also failed. The failure modes were identical to each other and resulted in a high-frequency but low-magnitude vibration due to the unbalance generated. These failures were somewhat puzzling since the impeller had successfully passed all other standard vibration tests.

Investigation

The failures that occurred were identified as fatigue which initiated at the junction of the fan blade to the rim. Resonance surveys were conducted on the impeller which indicated potential resonant modes within the operating rpm range of the helicopter. A dynamic strain survey was then conducted to evaluate the magnitude of stresses at the failure origin over its full operating speed range.

The result of the stress measurements is shown in Figure 9. It was evident that the stresses were exceeding the endurance limit due to a resonant amplification of the stresses at about 240 rotor rpm. This speed, however, is considerably above the normal operating speed of 225 rpm. The helicopter had been flying at the higher rotor speed primarily for the acquisition of performance data and the fan had therefore been operating at significantly higher stress levels.

A finite-element analysis of the impeller was conducted in order to establish resonant frequencies and mode shapes. Excellent correlation was obtained between the analysis and the resonance survey testing. The model thereby identified a means by which design modifications could be made.

It should be noted that the presence of a resonance within the operating range does not necessarily mean that a failure of that component will occur; a substantial excitation source at that frequency is usually required. In this particular case the excitation source was the spiral bevel input pinion/gear mesh at mesh frequency (3,400 Hz). Since the impeller was mounted directly on the pinion shaft, the normal attenuation that would occur with a slender, torsionally soft, drive quill shaft was not present.

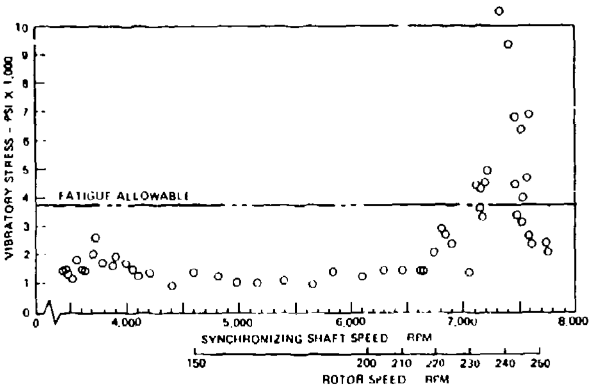

Figure 9. Stress Response During Speed Sweep on Original Fan Configuration

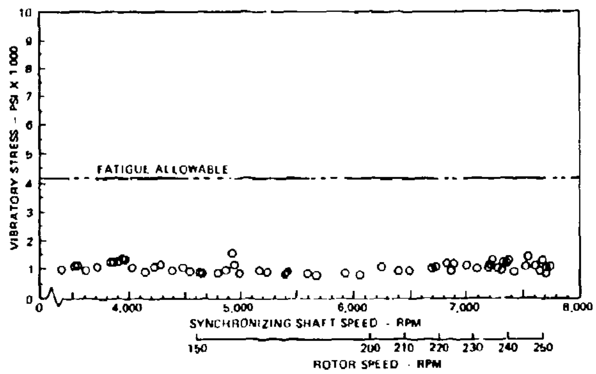

Figure 10. Stress Response During Speed Sweep on Modified Fan Configuration

Solution

The finite-element analysis indicated that by shortening the impeller blades by only 10 percent the critical resonant frequency was increased until it was considerably above the operating range. The modifications were made and the dynamic strain survey was repeated. The results of the modification resulted in a reduction of stresses well below the endurance limit, as shown in Figure 10.

Although the shortening of the impeller blades resulted in a slight reduction of cooling capability, it was adequate for the initial development aircraft. The fan vendor redesigned the impeller to increase the cooling capability to the original design requirement while insuring a resonance-free operation within and beyond the design operating range.

Commentary

This represented a design application where finite-element analyses were used to accurately establish on a complex drive system component the resonances causing failure and then to provide a solution. Although vibration qualification criteria such as MIL-STD-810C exist and are used on Army helicopters, the actual environment (input excitation) is not known until after the hardware is built and tested. The compatibility requirements between the airframe manufacturer and the accessory vendor would have to be based on experience obtained on similar installations. It should also be noted that the military specification qualification requirements do not cover excitations for gear mesh frequencies greater than 2,000 Hertz.

ENGINE/AIRFRAME COMPATIBILITY

Situation

In 1970, shortly after the introduction of the T55-L-11 and -11A engines into the CH-47 helicopter, field service reports indicated an increase in engine, engine component, and engine mount-related problems.

Investigation

The CH-47/T55 engine installation is basically hard-mounted. It employs two front mounts on a yoke at the engine inlet housing and an aft vertical support link at the engine diffuser flange as shown in Figure 11. The outboard yoke-to-airframe point is connected by a drag strut to take out high fore-and-aft maneuver loads. Engine vibration had rarely been a problem on the CH-47A and B models with this type of installation and with the T55-L-7 engine.

An extensive in-flight engine and aircraft strain survey conducted in 1970 provided a wealth of information on CH-47C engine/airframe interface dynamic characteristics. In particular, the survey identified rotor 3/rev as the predominant excitation frequency in the engine mounting system. Inlet housing stresses and drag strut loads were also shown to increase significantly with changes in rotor speed, thereby indicating a resonance. With these findings a ground shake test was conducted to determine the resonant mode being excited by rotor 3/rev.

The shake test revealed a 14.2-Hertz rigid-body yaw mode. Installation of the -11A engine (40 pounds heavier) dropped the modal frequency by 0.4 Hertz and, together with a rotor rpm increase from the A to the C models, resulted in operating in closer proximity to the resonance. The net effect was an increase in 3/rev inlet housing stresses. Additional testing showed that reducing the drag strut bolt torque could lower the engine yaw mode frequency into the CH-47C operating range (11.5 to 12.5 Hertz). Complete elimination of the drag strut lowered the mode to 7.5 Hertz, well below the CH-47C operating range. Removal of the drag strut, however, was not a practical solution since it was needed to assure acceptable cross-shaft alignment under high maneuver g's and jet thrust loads.

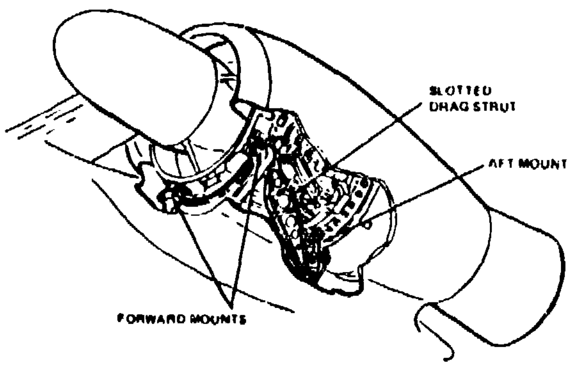

Figure 11. CH-47C T55 Engine Installation

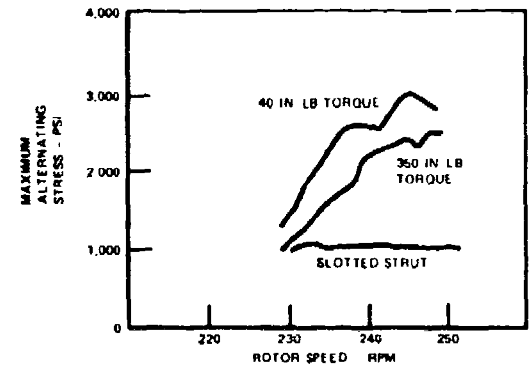

Figure 12. Engine Inlet Housing Stress at 140 Knots

Solution

A flight test was conducted with a slotted drag strut that is, the strut was free to move without restraint through normal operating displacements, but it bottomed under higher maneuver loads. The result was a substantial reduction in engine inlet housing 3/rev stresses, as shown in Figure 12. The slotted drag strut did not adversely affect stress due to other prime rotor harmonics or produce any significant 3/rev response during the engine starting procedure. No rise in stresses occurred when the strut bottomed. This was therefore felt to be an adequate solution. Experience has shown this to be a proper solution since no similar failures have occurred since the slotted strut was incorporated.

Commentary

Analytical efforts conducted at the time to predict installation dynamic characteristics met with only limited success. Manual calculations for fuselage stiffnesses were used initially and finite-element model analyses were used later. The accuracy of the analyses, however, was found to depend significantly on model idealization and on end-condition assumptions, and therefore very detailed modeling was required.

In this case engine bending was not a contributing factor. In installations where it is a factor, close coordination between engine and airframe manufacturers is required. Frequency modal analyses can easily be conducted with current technology through detailed finite-element models generated with close cooperation between the engine and airframe manufacturers.

CONCLUSIONS

It is evident from the few examples cited that there are many variations on the engine/drive/airframe compatibility theme. It is not always possible to establish cause-and-effect relationships. However, through a combination of analysis and testing, proper compatibility is usually achieved. Some specific conclusions based on the examples presented are:

1. Analysis and/or rotor/drive/engine system integration testing is required to establish optimum starting, operating, and braking procedures.

2. Airframe panel excitations due to fluctuating aerodynamic pressures at rotor n/rev frequencies should be evaluated with a finite-element model of the entire airframe structure in order to establish the resonant frequencies and the forced-response stresses.

3. Helicopter engine/drive system torsional instability can be prevented if care is taken to accurately represent both engine and rotor systems in the torsional analysis. Particular attention should be paid to proper lag damper representation, including hydraulic compressibility effects.

4. Self-excited vibrations of transmission-mounted components can be minimized by using finite-element analytical methods in the design stage and by test verification of the initially received hardware. Analysis and testing should not be limited to rotor per rev and shaft per rev, but should include gear tooth mesh frequency excitations as well.

5. Accurate finite-element method analyses and/or shake testing of all engine installations, whether hard-mounted, damped, or isolated, are required to determine potential vibration and stress problem areas. It is desirable to formulate engine/airframe manufacturer interface agreements during the early design stages in order to obtain a timely resolution of the potential interface dynamic problems.

6. It should be evident from this review that many of the problems discussed occurred many years ago (as much as 16). Because of these encounters the need for compatibility analyses between various systems was established and has therefore become a way of life with today's helicopters. A combination of advancements in technology and improved communications between prime and subcontractors has significantly reduced the recurrence of such problems during the past several years.

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PROBLEMS OF ENGINE RESPONSE DURING TRANSIENT MANEUVERS

by

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SUMMARY

Helicopter transition flight regime and extreme flight maneuvers determine abrupt variation of the rotor drag torque during each revolution. The corresponding effects on the engine are particularly treated in the present paper.

Rotor rotational speed fluctuations are found to be less than plus or minus ten percent. Repeated surging and the attendent transient torsional loads from the engine can cause damage to the airframe components.

The angular rotor acceleration, cyclically variable, is also influencing the turbine gas producer rotor speed, introducing flow distortions and aeromechanical effects in all the engine. It is found that periodic and inertial blade loading may have serious consequences with respect to discs and shaft to which the blades are attached. Flutter may by expected to exert oscillatory blade root reactions. The higher harmonics of the excitation over the discs may provoke relevant flexural modes of forced vibration.

Operation in transient rating with pitch increase or decrease has the greatest effect on the helicopter flight performance, owing to the power application capability and the fuel control system adaptability. It is therefore shown that, for having sufficient engine-airframe dynamic compatibility, aircraft developments have to incorporate some very significant technical advances, including harmonic integrated controls for propulsion and flight systems.

INTRODUCTION

The rapid trim change determined by the pilot during helicopter transient maneuvers is quite often interfering on the engine, in such a way to give inadequate fuel feeding. Moreover, the rotor blade aeroelasticity, as result of cyclically variable airloads, may introduce alternating stresses and fatigue effects on engine components. Because of the rotor-fuselage integration, aeroelastic instabilities in rotorcraft have to be predicted for a more reliable engine control design, Ref. 1.

The fundamental limitation on helicopter maneuverability is the rotor's maximum equilibrium thrust capability, as instantaneously delivered by gas turbine engines running at rpm much higher than the main rotor.
Fatigue limits imposed on helicopter engines are more severe than in other automotive applications, because of the vibration level induced in the engine core. Periodic aerodynamic and inertial blade loadings may have serious consequences with respect to the shafts and the discs to which these blades are attached. Flutter, with either random or uniform phasing between adjacent blades, may exert oscillatory root reactions that integrate for the entire disc and excite a shaft resonance.

Significant transient loadings of engine structure itself are associated with rotor blade torsional and flap-lag oscillations, figure 1. Moreover g forces from helicopter turns, pullups, hard landings, etc., act on the lifting and thrusting rotor and cause displacement of the rotating parts of the engine, as consequence of accelerations as high as 10 g in some maneuvers. Large gyroscopic moments are created when a spinning body such as the turbo-engine rotates about some axis other than its spin axis. The size of these moments is a function of the body spin speed, mass and distribution of mass of the spinning parts about their spin axes (as transmission systems with roller gear reduction unit), and the helicopter pitch/yaw rate. The resultant forces cause damaging cyclic bending of the rotating disks, blades and shafts. Gyroscopic moments of military helicopters may be of the order of 270 m - kN. It should be noted that gyroscopic forces are generated not only by helicopter maneuvers but can also be induced on engines. With the per cent occurrence of the standard maneuvers load spectrum, the calculated lifetimes may be more than 5 000 hours for the rotor shaft, 11 000 hours for the rotor hub and 22 000 hours for the blades, and much less for the gas turbine engine.

MANEUVERABILITY AND ITS SIGNIFICANCE FOR ENGINE OPERATION

Considering maneuverability, the load factor capability is most important. The limits are: engine power in a steady turn, and rotor stall in a deceleration turn. For future helicopters, g levels over 3, up to high speeds, will be required.
Maneuverability is probably one of the most important requirements in military helicopter operations. It is largely dependent on the type of rotor system used. A high degree of maneuverability can only be obtained with a rigid rotor, figure 2. In fact, for hingeless rotors without flapping hinges, it is possi-

ble to transfer high moments from the blades to the hub and the fuselage.

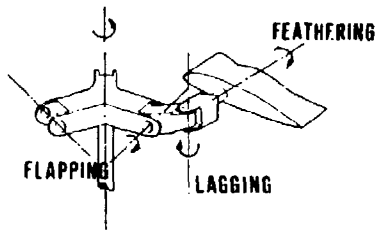


Fig. 1 - Typical hinge arrangement.

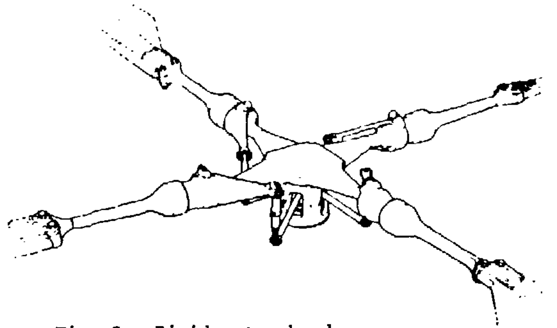
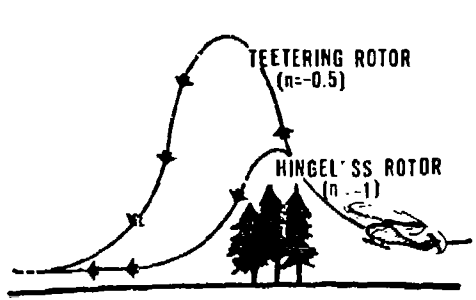


Fig. 2 - Rigid rotor head.

While the relationship between maneuverability and the functional mechanism is well understood today, as in terrain avoidance maneuvers and nap-of-the-earth flights in typical antitank missions, quantification still poses some problems. Frequent and abrupt changes in speed place severe stresses on the engines, gears, and all dynamic parts, which are reflected in special engine requirements, as in a terrain avoidance compound maneuver. From positive acceleration, as large as + 2 g, during pull-up, the three dimensional mission is here going on through negative acceleration (sometimes in between - 1 g and - 0.5 g) in the following push-over/rolling maneuver, figure 8. The practical consequences of such maneuvers are shown in figure 3 with the comparison of the helicopter exposure envelope for a hingeless and a teetering rotor (figure 4). The reasons for the different behaviour of the teetering and the rigid rotor are found in the different control moment capabilities of the two systems and the corresponding steering reactions.
The control helicopters with articulated rotors is mainly done by inclination of the thrust vector S, figure 5, thus producing a moment around the center of gravity of the vehicle. For a helicopter with a hingeless rotor system, an inclination of the thrust vector is combined with a strong hub moment, and the moment around the center of gravity is a combination of this hub moment and the moment due to the thrust inclination. The loading of the rotor shaft and the gearbox with its suspension is different in the two cases. For the articulated rotor the moment is built up linearly toward the center of gravity; whereas for the hingeless rotor the hub and the shaft are subjected to a relatively high moment loading. Trim conditions, which need a rotor produced moment to overcome the travelling center of gravity, require an alternating first harmonic moment in the rotating system for the hingeless rotor. Because of the equivalence of cyclic control and blade flapping, only an inclination of the thrust would be necessary. Higher harmonic blade loads resulting from the flow conditions in forward flight produce alternating forces at the hub for both rotor systems, and, in addition, for the hingeless rotor, moments at the rotor root and the hub. The feature of the hingeless rotor of transferring high moments from the rotor-blades to the hub and the fuselage results in a changed situation for the control characteristics. The control of the helicopter becomes more powerful, faster and more direct, and nearly independent of rotor thrust.
As in figure 5, the control moment around the center of gravity of the helicopter is only produced by the tilt of the thrust vector, in the case of central flapping hinges with teetering articulated rotors. This moment is relatively small, but an additional rotor moment is produced by the shear-forces at the flapping hinges. This rotor moment is much higher (about five times that of an articulated rotor) in the case of the hingeless rotor, which can be considered equivalent to an articulated rotor with a flapping hinge offset of about 15 percent.



Altitude over cover ∿ 50 m for teetering rotor
∿ 15 m for hingeless rotor

Exposure time ∿ 9 s for teetering rotor
∿ 25 s for hingeless rotor

Fig. 3 - Helicopter terrain avoidance maneuver.

Military helicopters require a maneuver capability beyond a hover out-of-ground-effect condition. A power/maneuver margin equivalent to a vertical rate of climb of 150 mpm (meters per minute) was established as the minimum acceptable reserve. But higher turbulence levels are encountered with vertical component of 300 mpm, as in advanced attack helicopters. Typical mission requirements for military attack and tactical transport helicopters are: operation in confined areas; vertical or near vertical takeoffs and landings; downwind takeoffs and landings, which increases

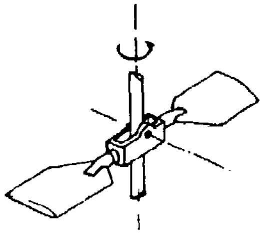


Fig. 4 - Teetering rotor.

demand for lift and power; reserve power to arrest high sink speeds and zero out airspeed at the landing zone; power requirement similar to vertical climb criteria for successful aborted landing.

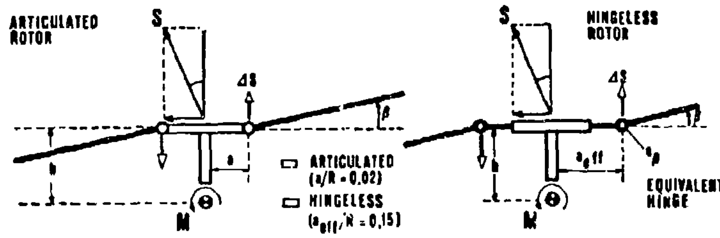


Fig. 5 - Rotor control moment capacity.

These technical requirements are determining the configurations of future military helicopters, i.e.: highest possible maneuverability, also in negative acceleration flight envelope; dual engine concept; performance reserves at all possible conditions; dual redundancy or fail safe concept of all important components. In the tactical environment, rapid and extreme maneuvers will be necessary, with: maximum load factors up to 2.5 g; roll angles up 1.4 rad; rolling speeds up to 0.9 rad/s; pitch angles up to 0.7 rad; pitching speeds up to 0.7 rad/s.
The rapid change of these parameters up to their extremes will, of course, result in high loads. Drive systems are designed to use 120% of the current power available at the design condition of 1 200 m/308 K. In addition, these drive system design criteria allow more efficient use to the higher engine power available at low altitudes and/or low temperatures, where heavy weight alternate missions are required.

Nearly half of all accidents can be attributed to factors related to inadequate performance or control capability. Nearly 25% of the total are related to inadequate design features which are being corrected or enhanced in the new design, such as: dual engines with single engine inflight capability, to minimize accidents attributed to engine failure; advanced materials technology and redundant design features to help eliminate material failures; instrument flight capability to avoid weather related accidents; enhanced damage tolerance features to minimize effects of foreign object damage to engine and aircraft structure.

One of the critical mission that seizes the helicopter in the preliminary design stage is the high-speed flight requirement at sea level and 3 000 m density altitude with an adequate performance envelope, figure 6.

Peak engine power is required in maneuverability to make rapid changes in flight path and altitude under the precise control of the pilot, as, for instance, in hovering maneuvers (turns, jump take-offs and quick descents), accelerations and decelerations, and target acquisition. Load factor varies over a wide range through the maneuvers.
A typical turn maneuvers is described on figure 7. In order to accomplish turn maneuvers, an analytic autopilot is required to present the pilot flight-path control in executing the maneuver with load factor as high as 2.0. After a load factor build up time, it follows a maneuvers execution time of 3 seconds during which the load factor equals or exceeds the specified level. For instance, the terrain-avoidance maneuver requires pull-up to achieve 1.75 g's at 150 knots within 1.0 second, sustain 1.75 g's for 3.0 seconds, push-over to achieve 0.0 g for 1.0 second.
The MBB-BO 105 hingeless rotor helicopter, for instance, offers the potential, as other modern helicopters, for acrobatic loops, figure 9.

The propulsion system is often responsible to fail because of its original deficiency in satisfying instantly excessive loading factors imposed on the cyclically variable rotor torque during the pilot transitional inputs from a flight segment to another. The computerized fuel control is effectively inadequate to establish continuous equilibrium between shaft drag torque and delivered engine torque, in that resulting excessive rpm inertial dissipation.
Moreover, limitations are imposed up on the operations of helicopters as a result of the high aero-

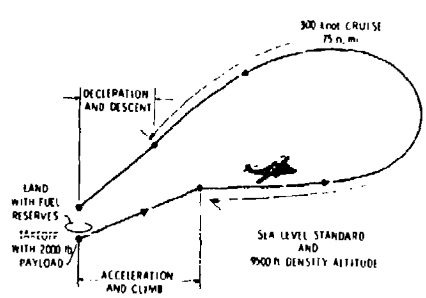


Fig. 6 - High speed requirement.

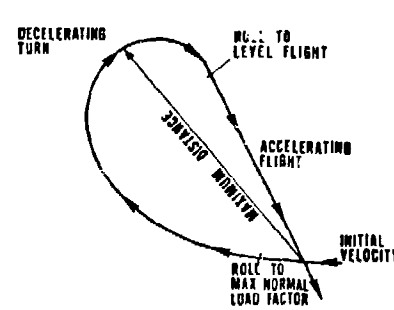


Fig. 7 - Typical turn maneuver.

elastic vibration levels encountered in flight, and transmitted to the driving engines. The associated periodic loads on the engine rotating parts, particularly during the so-called transition flight regime between hovering and forward flight and during high-speed flight, are contributing substantially to material fatigue and consequential high rate of replacement and maintenance of component parts. Extreme flight conditions are right and left turns, rolling pull-outs, longitudinal reversal, cyclic and collective pull-ups, slope landings and starts.
Output shaft rpm cannot generally vary more than 10%, and gas turbine engine operation may become critical at the minimum rpm values, with a consequential loss of rotor thrust.
Surge still exists in engines that have been abused. Repeated surging and the attendant transient torsional loads from the engine can cause damage to the airframe components.
The interstage bleed system automatically relieves the compressor of a small amount of air, during the period of the engine acceleration cycle, at slight loss of power. The entire sequence operation should be controlled by the fuel control which should sense gas producer rotating speed, fuel flow and pilot demand, therefore ensuring proper opening and closing of the interstage air-bleed.
The fuel control is a hydro-mechanical device, with fuel regulator and power turbine governor. The acceleration and deceleration fuel flow control may be in excess of the engine's ability to immediately produce the desired power. To get a good performance, the amount of fuel added to the air in the combustor must be exact at all times, in acceleration, deceleration and steady state engine operation.

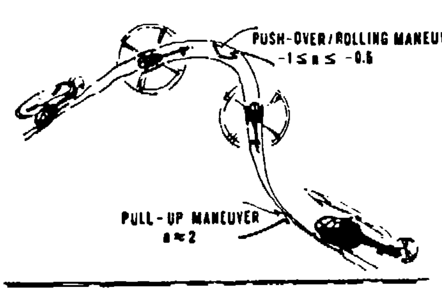


Fig. 8 - Typical terrain avoidance compound maneuver.

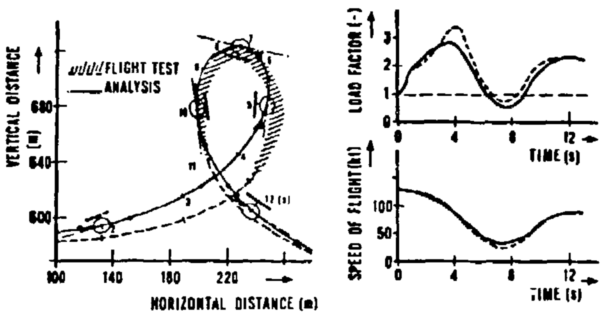


Fig. 9 - Potential acrobatic power.

As a result, the engine should be brought through transitory condition as rapidly as possible, keeping the rotorcraft out of dangerous mixtures. Boost pumps, check, metering, dump, pressurizing, and shutoff valves, and pressure regulator, of the automatic fuel flow system, are a very complex matter, susceptible of possible malfunctioning failure during rapidly changing operation. The emergency fuel system must be operated with some care to avoid engine damage. Gas producer governor and power turbine governor are sensing any deviation in steady state of the corresponding compressor-turbine rotor and free turbine rotor. The transient air-bleed control on the compressor rotor of the gas producer turbines is not able to operate correctly during power turbine rotor acceleration. Unsteady airloads on the helicopter main rotor might be so high to stop at all the combustion process.

INFLUENCE OF TURBULENCE ON ENGINE RESPONSE DURING MANEUVERS

The rapid development of helicopters with higher disc loadings and higher forward speeds have made the gust loading problem more significant. A changed loading situation as well as accelerations and motions of the helicopter are caused by atmospheric turbulence; the effects on structural loads are important from the design point of view. From the other hand, the analysis of atmospheric turbulence shows that it contains energy at all frequencies, from hundreds of hertz down to fractions of a hertz; the engine response to turbulence during maneuvers might be aerodynamic excitation, with the natural frequency of the fundamental component of blade excitation, and critical resonances.
Turbulence is known be a problem close to the ground, particularly in the final stage of the landing approach, and in the terrain avoidance compound maneuver. One relevant phenomenon is the cold autflow of air from a thunderstorm forming a squall front, which can cause a very rapid change in airspeed, and hence in rotor blade lift-force and in engine operation.

The main response of . helicopter structure to severe turbulence is a succession of differing magnitude jolts of normal acceleration, or a single jolt. The response to a give size of gust will vary from helicopter to helicopter, depending on rotor disc-loading, speed and lift-curve slope. If the disturbance is of sufficiently large magnitude, it can result in structural damage to the helicopter.
The engine behaviour is influenced by the gust sensitivity regarding moments and load factor, even if designed to mitigate their effects.

At high speeds the load factors in maneuvers with sharp edged gust impact are relatively high, and the effect on the engines become critical. Significant transient loading of compressor structure would usually be associated with a gross change in the throughflow, that is either truly aperiodic (or of very low frequency). The problem of sudden changes in throughflow is closely related to the aeroelastic prob-

lem of a single airfoil passing through a sharp-edge gust front in the atmosphere, such as might be produced by a remotely originated blast wave. But, the aerodynamic moment resulting from aperiodic forcing, and the corresponding torsional blade or vane response, are not important when the disturbance is truly aperiodic, since the vibration would soon damp out due to aerodynamic and other forms of damping. However, if the compressor is operating in a cyclic surging or stalling condition, the required number of cycles for fatigue and consequent failure could be accumulated in a short period of time. Low cycle fatigue might so be encountered in compressor rotor blades, nowadays subjected to exceedingly high centrifugal stresses. The response of rotors and shafts, very stiff in the axial direction and at high natural frequencies in their longitudinal modes, is nevertheless usually quite safe.

Distorted flow into a helicopter gas turbine, as consequence of turbulence in maneuvers, can trigger engine instabilities. The axial flow compressor is particularly sensitive to flow non-uniformities. Compressor stall can lead to engine surge. This condition may be dangerous depending on the particular propulsion installation and engine, and on the vehicle maneuver, because the source of the flow irregularities is the inlet.
Among the propulsion system margins are buzz or inlet unstart, compressor stall, turbine inlet temperature, and mixture ratio limits to maintain combustor flame stability.
Violent flight maneuvers can cause high levels of distortion. Propulsion stability depends on Mach number, Reynolds number, angle of attack, angle of yaw, and power setting. Figure 10 illustrates regions of propulsion instability in the velocity (V) - load factor (n) plane. At low velocities, very large angles of attack are necessary to generate helicopter lift. Large angle of attack adversely influences propulsion stability. At higher velocities, the angle of attack is less for a particular load factor as a result of increased dynamic pressure. Unfortunately, the increased dynamic pressure also means increased distortion levels.
The direct effect of inlet distortion on aeromechanical response is not the only one to be considered. Indirect effects of severe inlet distortion are also producing potentially harmful aerodynamic environments for blading at operating conditions. Examples of such indirect effects are the ones arising from the thick annulus boundary layer and the degradation of flow quality. The first one may stall the compressor rotor blade tips and give rise to a rotating stall pattern of limited radial extent, giving rise to periodically reversing throughflow and surge (in the case of massflow reversal); the sudden forward travelling of the compressor discharge pressure, coexisting with the flow reversal, implies a pressure wave, the so-called hammershock, going through the compressor into the inlet. The second one may induce unusually high turbulence levels and thick wakes on the stator and rotor airfoils, producing negative incidence in some stages of the axial compressor.

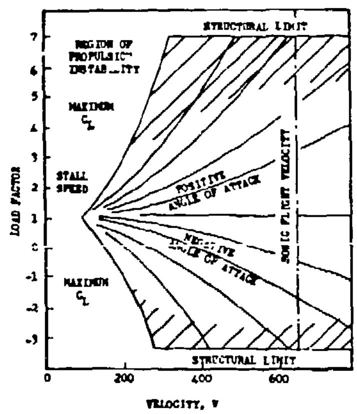


Fig. 10 - Regions of instability.

The periodic forcing on a rotor blade operating in a circumferential distortion is, however, the most significant aeromechanical response, because of the multi-term harmonic inflow conditions. For certain combinations of frequency, cascade geometry and relative Mach number, very close to aerodynamic resonance, rather large magnitudes of rotor blade lift and moment may be obtained.

The degradation of flow in an axial turbomachine may lead to the onset of self excited blade vibration (stall flutter) or self-excited fluid oscillation (stall propagation).

When wind blows over the ground, a boundary layer develops. This layer may extend 30 meters or more above the ground. As with all boundary layers, the flow is rotational. A turbojet pumping air from the boundary layer concentrates the vorticity. This is analogous to the vortex in a draining bath tub; the vorticity present in the bath water is concentrated as the tub drains. The vorticity entering the compressor can affect the stall margin in an adverse manner.

In the flight maneuver environment, a helicopter turbojet must contend with internal perturbations caused by unusual flow at the inlet. For that, the axial flow compressor may be required to operate at flow rates and pressure ratios other than the flow rate and pressure ratio for which it is designed.
In addition, transient and unsteady effects occur at air intakes, with pressure fluctuations due to unsteadiness of shock waves, expecially during sharp maneuvers at large variations of the angles of attack and yaw.
Flow distortions may be even so large to stop or seriously damage the turbojet engine.

INFLUENCE OF MAIN ROTOR AEROELASTICITY ON ENGINE RESPONSE DURING MANEUVERS

In maneuvering flight, where sharp turn and abrupt pull-ups may be required, severe blade stall can be encountered, because of rotor rpm decay or high load factors. Recovery in these cases is much more difficult, if the rotor is quite high loaded. Reduction of collective pitch and/or increasing rotor (and engine) speed will help.
A chart, or better a computer, is utilized for the prediction of blade stall. The pilot is however expected to know and avoid the general conditions conducive to the stall condition. The important thing to

remember is that blade tip stall is not solely associated with high helicopter velocities. It can occur, under certain conditions, at surprisingly low forward speed.

There are numerous unsteady aerodynamic events that may occur in a single revolution of a helicopter rotor system operating at moderate to high forward flight. The severity of the dynamic airloads associated with the unsteady aerodynamics depends upon the specific operating condition. Some of the more severe operating conditions are: maximum forward speed in level flight, maneuvering flight, rapid descents, and flared landings in ground effect.
The pratical limitations of rotor operation are analogous to turbine/inlet matching aerodynamics, where the fundamental problem is one of dealing with extreme, time varying, three-dimensional distortions of the rotor inflow distribution.
The transient Mach number effects on the advancing blade include transient lift, drag, and pitching moments associated with the rapid movement of the shock wave over the upper surface of the blade. The interaction of the shock wave and boundary layer can also induce transient shock stall.

Helicopter maneuvering flight is severely limited by unsteady aerodynamics, because of rapid increase in magnitude of the rotor oscillatory control loads and overall airframe vibration levels.
The important factors in the maneuvering flight unsteady airloads are wake geometry, tip-vortex flow, blade-vortex interaction, and dynamic stall.

It is quite complex to simulate analytically flight maneuver loads on helicpter propulsion systems, considered as a whole from the rotor to the engine. High-performance gas turbine engines are sensitive to not only self-induced cyclically changing loads but also to extremely induced loads which result from helicopter maneuvers. These external loads cannot be adequately simulated for prediction of the complex force interactions; and experimental data, obtained from flight tests, are not sufficiently definitive.

The engine is expected to operate in a very hostile environment, being exposed to extremes of temperatures, pressure, vibration, and mechanical forces within the engine. In addition, the engine is exposed to a variety of accelerations in all directions, thus imposing large inertial and gyroscopic forces on an already complex loading environment. The engine is expected to function under all conditions with small clearances between rapidly moving and stationary parts and to exhibit little degradation in performance after long period of operation.
As it is known, a typical flight procedure consists of a check of output torque as a function of gas generator speed against installed performance.

An engine stall is especially bad in a helicopter bacause of the rapid load reversal which occur in the rotor drive system. A stall is indicated by a sudden drop in compressor discharge pressure accompanied by an increase in exhaust gas temperature.

Forward speed is limited by blade stall on the retreating side of the rotor and by compressibility on the advancing side of the rotor. In order to keep roll moments balanced on a conventional single rotor, the lift potential that can be developed on the advancing side of the rotor disc is limited by that which can be developed on the retreating side. That can be seen on figure 11 for a conventional two-blade single rotor. Figure 11 compares, for a hingeless helicopter rotor system, the lift distribution of a conventional single rotor and coaxial counterrotating rigid rotor. The stiff blades, largely eliminate classical retreating blade stall and permit operation at higher thrust coefficients and advance ratios. Moreover, the stiff blades are able to operate more easily without aeroelastic instabilities as rotor speed is reduced. Such operation is required for high-speed flight where the advancing blade tip Mach number should not exceed a value of approximately 0.85.
In the case of moderate helicopter advance ratio (forward speed/blade tip speed) and of local incidences lower than that of profile stall, the azimuthal evolution of the blade local loads (for a rigid three blade rotor) are presented on figure 12 for different blade sections (at radius r in respect to the tip radius R).

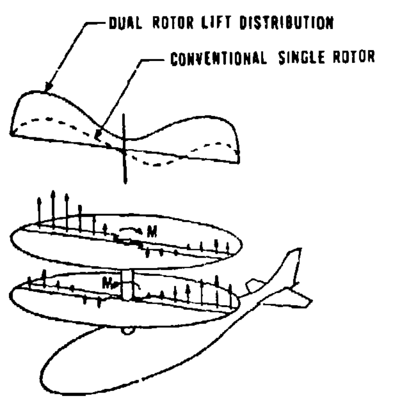


Fig 11 - Rotor lift distribution for coaxial counterrotating dual rotor.

The complex combination of external and internal operating forces and maneuver forces presents a big obstacle to determining the distortions occurring in the engine during helicopter flight. Because of rotor movement, case bending and ovalization, rotor shaft and blade bending, blade extensions, and thermal stresses, it is difficult to predict the relative movement of engine components under all flight maneuvers. Abnormal vibrations in an engine installation may be caused by malfunctioning engine-mounted airframe accessories, engine mounts and other external connections, as well as by impulsive flow phenomena caused by blade-vortex interaction on the advancing side of the rotor disk. External airloads may be so highly unsteady to produce engine instabilities, going from the accompanying the distinctive "blade slap" acoustic signature of some helicopter rotors to those determining engine fatigue, failures and stop.

The need for accuracy in calculation of engine-rotor performance can be emphasized by noting that a 1% change in the lifting capability of a rotor, at a given rotor input, may mean a 10% change in payload for the helicopter. From the other hand, accuracy in performance calculations is required because the engine could be not able to support instabilities due to unsteady airloads on the helicopter rotor.

With the development of helicopters capable of higher flight speeds, the problem of blade stall flutter has become one of major importance. Classical flutter involves coupling between two or more natural modes of vibration. In stall flutter, the flutter frequency tends to become equal to the natural torsional frequency. This implies that in this case the torsional (or pitching) motion predominates.
Large high-frequency oscillations in torsional moment (TM), lift and aerodynamic moment coefficients, on the retreating side were detected in flight-test data taken from a rotor blade during a maneuver, as shown in the plot of figure 13. This response was the result of dynamic stall induced by previously formed tip vortices which, under that particular maneuver flight conditions, pass under the blade at the azimuth positions indicated in figure 13.
High vibration levels are established by aerodynamic loading, and aeroelasticity of the rotor blades, and resonance of the structure. Increased periodic forces are occurring when the rotor thrust is increased, during transition from hovering to forward flight, and at higher tip-speed ratios. The harmonics above the fourth are generally below 20 percent of the principal harmonic force. In turn, the magnitude of the harmonic forces other than the nth one for n-blade rotor in generally below 50 percent of the principal n-th harmonic.
The analysis of blade torque moments indicates the magnitude of the harmonic components to be about 4 ± 5 percent, respectively, of the steady components of the moments.

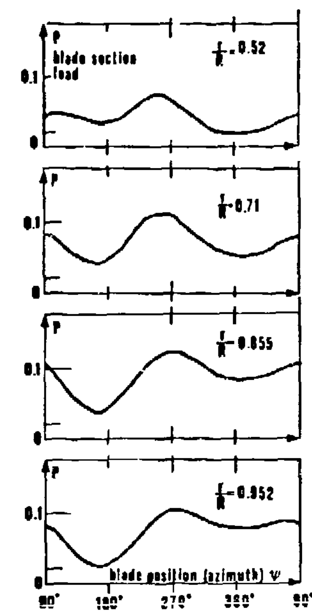


Fig. 12 - Blade local load for a rigid three blade rotor.

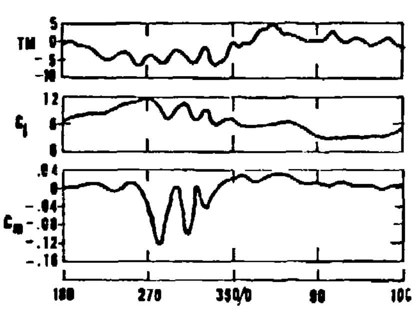


Fig. 13 - Measured time history of section loading and moment coefficient and blade torsional response.

But, still now, rotorcraft are operated at reduced speeds, as much as 20% below the forward velocity that they might otherwise achieve, because of vibratory environments. However, the understanding the sources of vibration is more and more required for reaching forward speed as high as 370 km/h.

EFFECTS ON THE ENGINE OF HIGH SPEED FORWARD FLIGHT AND ABRUPT MANEUVERS

In order to keep the helicopter airborne, static or at a given flight speed, the main rotor must produce thrust, by giving a momentum to a mass of air. In so doing, power must be spent to produce acceleration of the air.

In this condition, the engine must supply power to overcome the main rotor induced and profile drag, and power to turn the tail rotor and surmount friction in the transmission and drive system, and rotor losses. Profile drag power increases only slightly as normal forward speed is increased; but, at very high forward speeds, and as blade stall or stall flutter is entered, it mounts rapidly. Required parasite power, varying as the cube of the airspeed, increases very rapidly at higher speeds and accelerations. At zero airspeed, the induced power requirement is quite high, because downwash velocity is near maximum value.
As illustrated in figure 14, for a helicopter with adequate power in normal flight, there is ample power available to permit hovering and climbing and to permit overlanding of the helicopter as airspeed is built up.

During high speed forward flight, the multi-blade rotor drag torque Q is variable during each revolution, because of:

- cyclic pitch control (by blade feathering) of the amount of forward thrust for each blade, according the flapping and lagging motion of articulated or hingeless rotors, with blade stall vibration and unsteady aerodynamic effects at very high flight speed;

- coupled flap-lag-torsional multi-blade aeroelastic effects, figure 15, stall flutter and air resonance;
- coupled rotor/fuselage aeroelastic feedback effect on the rotor.

Adimensional amplitude $\Delta Q/Q$ and angular velocity $\Omega/\Omega_{average}$ cyclic variations of the rotor torque may be, near the maximum acceptable flight speed, respectively, of the order of 0.1 and 0.05, as compared with the measured values of $\Delta C_\ell/C_{\ell\ av}$, $\Delta C_T/C_{T\ av}$ and $\Delta C_m/C_{m\ av}$ (respectively related to the lift, thrust and moment coefficients), as in figures 11, 12 and 13.

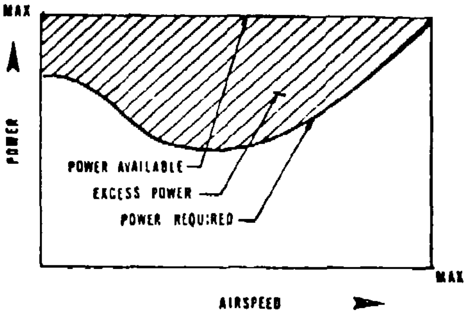


Fig. 14 - Excess power in respect to normal flight.

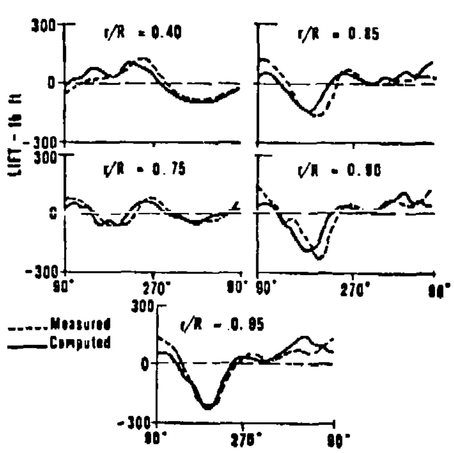


Fig. 15 - Computed and measured time history of blade airloads.

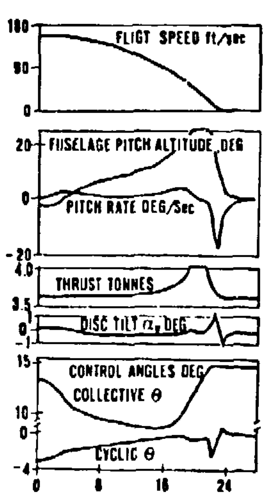


Fig. 16 - A deceleration maneuver time history.

During transition flight regimes and extreme flight maneuvers, as in the acceleration maneuver time history on figure 16, the rotor drag torque Q is varying abruptly during each revolution, because of:

- sudden airloads overimposed to the cyclic pitch control of the rotor thrust;
- vibration and blade flexibility effects, overimposed to stall flutter and flap-lag-torsional blade aeroelasticity;
- effects on the engine of the strongly aeroelastic system corresponding to the rotor-fuselage integration.

Such cyclically variable main rotor drag and induced torque Q is reduced in amplitude and disuniformity to the shaft of the free (power) turbine. The speed reduction ratio between engine and main rotor is ranging, for the present helicopters, from 70 to 80 : 1 and will be increased in the foreseeable future. About in the same ratio, the main rotor torque and frequency disuniformities are reduced to the gas turbine engine shaft. Fortunately, the rotor unsteadiness during a single revolution is spread in 70 ÷ 80 engine shaft revolutions!

The damping action of the transmission layout, from the main and tail rotors to the gas turbine engine, is necessary for a reliable continuous power availability. From the other hand, the lifed items in a helicopter transmission system, for example as complex as in figure 17, must be replaced, due to possible fatigue damage after expiry of a fixed flying time, typically a few thousands hours.
The reduction of gas turbine powerplant weight, in comparison to previous reciprocating engines, has made possible to install enough power for good performance. Unfortunately, the same kind of revolution has not taken place in the power transmission, where some components are operating at very high angular velocity and acceleration. Since both gas turbine and helicopter rotors operate at a tip speed based on Mach number, the rotational speed of the engines will increase with advanced technology engines, where the turbine inlet temperatures will increase appreciably. The speed reduction ratio will also increase.
Speed reduction has been accomplished in the past by putting a number of low-reduction-ratio units in series. As an example, the CH - 47 Chinook, figure 18, has an engine turbine speed of 14 500 rpm, which is reduced to 11 500 at the engine gearbox, then to 6 780 in the mixer box. The final gear-box has an overall 30.72 : 1 reduction ratio through a bevel gear and two planetary gear stages to a rotor speed of 222 rpm. The gas turbine engines that have been designed for present helicopter use, such as the General Electric T-58 on the Agusta Bell 204 AS, the Lycoming T-53 and the Allison T-63, were required to have an integral gear-box to reduce the output speeds to approximately 6 500 rpm, because it was believed this was as high a speed as could be conveniently handled by helicpters.

Transmissions for front and back engines are shown in figure 19 and 20. A free turbines is direct-

ly under the transmission in the solution of figure 21.
A design simplicity has been followed on the transmission system of the semi-rigid rotor Lynx (figure 2), as on figure 22. In particular, this solution allowed the use of a high reduction ratio in the final stage, permitting a major reduction of transmission components, especially in the high torque paths. Thus, the component numbers (7 gears and 19 bearings) are reduced from 44 (16 gears and 28 bearings) in the Sea King to 26 on the Lynx, with anticipated proportional benefits in reliability and rapid growth of fatigue life.

Inertia loadings of gears, shaft and bearings, of the transmission system may be serious consequences with respect to the attached gas turbine engine. A rich variety of vibration phenomena gives rise to distorted throughflow in the engine. Mechanical and aeromechanical excitations exert, in such a way, dangerous oscillatory root reactions on the blades, Ref. 3.

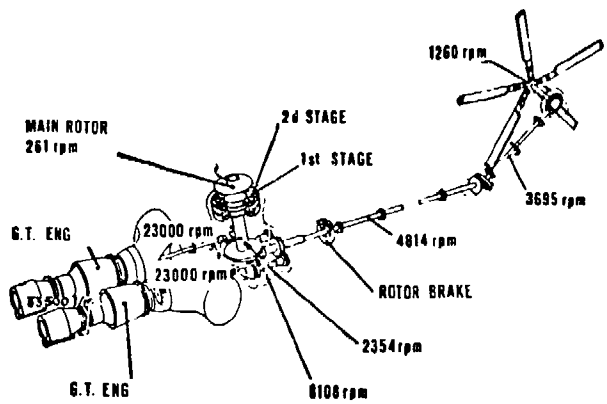


Fig. 17 - Helicopter transmission layout.

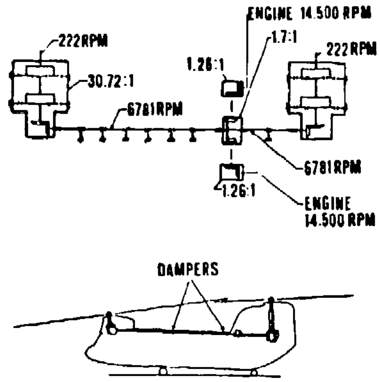


Fig. 18 - CH-47 Chinook transmission system.

When the rotor torque Q is reduced about 70 ÷ 80 times to the free (power) turbine shaft, its periodic disuniformity, created by moderate speed forward flight at limited vibrational and aeroelastic effects, is practically absorbed through the engine gear box (between power shaft and drive shaft) and the planetary gear unit (between drive shaft and rotor shaft), with passive heat production.
But, when the forward speed is so high to start blade stall vibration and large amplitude aeroelastic effects, the engine is quite largely influenced and cannot compensate the required power fluctuations through the automatic fuel control action. We may expect in this case considerable compressor-turbine and free (power) turbine vibration and engine flow distortion. In absence of a self acting automatic control, to avoid high flight speeds and too abrupt maneuvers, or in the case of pilot error, engine failures and/or flame-outs could be involved.

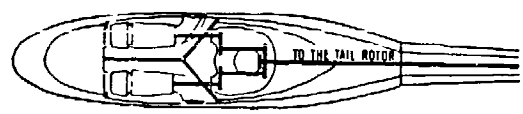


Fig. 19 - Front engine transmission.

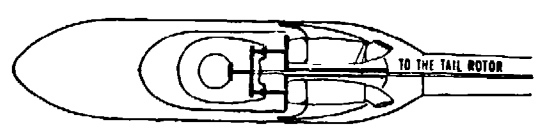


Fig. 20 - Back engine transmission.

With Q_e and ω_e corresponding to instantaneous values of the power turbine shaft torque and the shaft angular velocity, the power balance at each azimuthal rotor position of a given flight condition may be expressed

$$Q_e \cdot \omega_e \cdot \eta = Q \cdot \Omega \qquad (1)$$

where η and Ω are respectively, the mechanical efficiency, due to power dissipation through the reduction gears, and the main rotor angular velocity.
The corresponding engine torque (considering the average torque Q_{av} for such flight condition)

$$Q_e = Q \cdot \Omega/\omega_e \cdot \eta = (Q_{av} \pm \Delta Q)\Omega/\omega_e \cdot \eta \qquad (2)$$

is cyclically variable, depending upon the rotor torque and the changing mechanical behavior (expressed by the value of η).
At constant forward flight speed, the control fuel setting is such to deliver, with a fuel flow G, (N/s), (W), corresponding to the following mechanical balance

$$G \cdot K = (Q_{av} \cdot \Omega_{av} + Q_c \cdot \omega_{ec})/\eta_t \cdot \eta_m \cdot \eta \qquad (3)$$

where Q_c , ω_{ec} η_t and η_m , are torque and angular velocity of the compressor turbine, and thermal and mechanical efficiencies.
Now, the periodically variable power turbine rotational acceleration $d\omega_e/dt$ is depending upon the inability of the fuel control system to vary instantaneously the delivered power, and it may be expressed

$$\pm\ d\omega_e/dt = \Delta Q \cdot \Omega/\omega_e \cdot (J_e + J_1 + J_2 + J_3 + J_r) \qquad (4)$$

where: J_e is the power turbine rotor system moment of inertia, and J_1, J_2, J_3, J_r, are the moments of inertia reduced (trough the square rotation speed ratios $(\Omega/\omega_e)^2$) to the power turbine shaft, respectively, of engine box gear, drive shaft, planetary unit gear, and helicopter main rotor. (In a complete drive system, as on figure 17, it has to be taken into account the tail rotor torque disuniformity and the reduced moment of inertia of the relative drive components).
The rotational acceleration is cyclically variable according the expression of ΔQ and Ω. Practically, such acceleration is also influencing the gas producer turbine speed, introducing flow distortions and aeromechanical effects on all the engine.

The main rotor incremental torque ΔQ , in respect to the average torque Q_{av}, is determining the torque Q_c and the rotational acceleration $d\omega_e/dt$ of power turbine shaft approximately according eqs. (2) and (4), for each azimuthal angle of the main rotor.
Even though only experimental data and directly measured values could help to solve the problem, analytical approximate procedures may be as it follows.

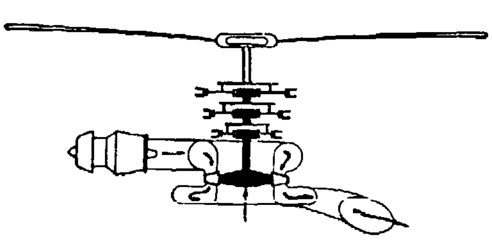


Fig. 21 - Transmission by free turbine.

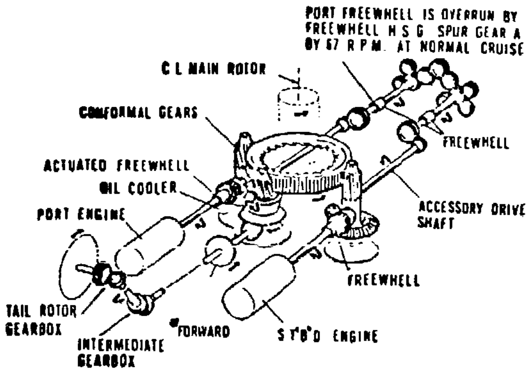


Fig. 22 - Semi-rigid rotor Lynx transmission.

Rotor torque in forward flight - With the blade assumed as a rigid beam with a flapping hinge, figure 23, the total induced air velocity W relative to a blade element distant r from the no-feathering axis (as resultant of the induced airpeed, the forward velocity V of the helicopter, and the rotational velocity $\Omega \cdot r$) has a chordwise component U_T (comprehensive of $\Omega \cdot r$)

$$U_T = - V \cos\alpha \sin\psi - \Omega \cdot r = - \Omega \cdot r(x + \mu \sin\psi) \qquad (5)$$

(with: advance ratio $\mu = V \cos\alpha/\Omega \cdot R$, and $x = r/R$) and a component U_P perpendicular to the chord having ignored the spanwise component).
In the region of the advancing blade ($0 < \psi < 180°$), the relative chordwise wind due to rotational speed is increased by a component of V ; in the region of the retreating blade ($180° < \psi < 360°$), the component of V reduces the relative wind. The wind is blowing from the trailing edge to the leading edge in the part of the retreating blade, where the forward speed component is greater than that due to the rotational speed, eq. (5). This happens, Ref. 1 and eq. (5), in a circular "reversed flow" region whose diameter is μ R, figure 23.
The contribution from this region is usually very small; for $\mu = 0.3$, the area of the reversed-flow region is only 2.5 per cent of the total rotor area (with very small velocities). But, when $\mu = 0.6$ and more (well above the top speeds of present-day helicopters), the reversed-flow region begins to assume importance.

The advance ratios are, for example, in current or developmental status helicopters:

- Agusta AB 204 AS (two-blade articulated rotor; rotor diameter, 14.63 m; cruise forward speed, 222 km/h; normal rotor rpm, 295-317; rotational blade tip speed, 226-243 m/s; cruise blade tip maximum forward speed, 288-305 m/sec, 1 037 - 1 098 km/h) $\mu = 0.253 - 0.273$;

- UTTAS (U.S. Utility Tactical Transport Aircraft System) (four-blade articulated rotor; rotor diameter, 16.36 m; cruise speed, 269-324 km/h; normal rotor rpm, 258; rotational blade tip speed, 221 m/s; cruise blade tip maximum forward speed, 296-311 m/s, 1 066 km/h - 1 120 km/h) $\mu = 0.338 - 0.407$;
- AAH (U.S. Advanced Attack Helicopter) (four-blade articulated rotor; rotor diameter, 14.63 m; cruise speed, 269-324 km/h; normal rotor rpm, 289; rotational blade tip speed, 221 m/s; cruise blade tip maximum forward speed, 296-311 m/s, 1 066 - 1 120 km/h) $\mu = 0.338 - 0.407$;
- ABC (U.S. Advanced Blade Concept) as the XH-59 A (three blade dual coaxial, counterrotating, hingeless rotor; rotor diameters, 11 m; maximum level flight speed, 297 km/h; maximum level flight speed with auxiliary propulsion, 519 km/h; normal rotors rpm, 333 (238 with auxiliary propulsion); rotational blade tip speed, 192 m/s (137 m/s with auxiliary propulsion); cruise blade tip maximum forward speed, 275 m/s, 990 km/h (281 m/s with auxiliary propulsion, 1 012 km/h) $\mu = 0.43$ (1.05 with auxiliary propulsion).

The torque dQ on a blade element $(c \cdot dr)$ is, figure 23, as function of the lift dL and the drag dD

$$dQ = r(dD \cdot \cos\varphi - dL \cdot \sin\varphi) \qquad (6)$$

It is, with $W^2 = U_T^2 + U_p^2 \simeq U_T^2$, $\cos\varphi \simeq 1$, $\sin\varphi \simeq U_p/U_T$, C_L and C_D corresponding to drag and lift coefficients

$$dQ = r(C_D - C_L\, U_p/U_T)\, \tfrac{1}{2}\rho\, U_T^2 \qquad (7)$$

where, the first term denotes the blade element torque dQ_d due to profile drag, and the second term corresponds to the blade element torque dQ_i due to rotor induced lift.
Eq. (7) becomes, in respect to the azimuthal angle ψ and the adimensional ratio $x = r/R$

$$dQ = dQ_d - dQ_i = \frac{x^3 R^4 \Omega^2 \rho c}{2} (x + \mu \sin\psi)^2 \cdot (C_D - C_L \cdot U_p/U_T)dx \qquad (8)$$

The rotor thrust dS on a blade element $(c \cdot dr)$ is, figure 23

$$dS = (dL \cos\varphi + dD \sin\varphi) \cos\beta \simeq dL \simeq \frac{x^2 R^3 \Omega^2 \rho c}{2 U_T} (x + \mu \sin\psi)^2\, C_L\, U_p\, dx \quad (9)$$

The values of the lift and drag coefficients in eqs. (8) and (9) may be deduced, with simple ideas of induced velocity calculated on the assumption of the rotor regarded as a lifting surface, in function of the local angle of attack $\alpha = \theta + \varphi$. The angles θ and φ, for each blade azimuth ψ, are depending upon the blade pitch "collective and cyclic" control (pitch control by feathering, figure 1, contemporarely all the blades and individually each blade) and the amplitude of the total induced velocity W (see the induced velocity distribution according to Mangler's theory, Ref. 4).
For instance, figure 24 shows that the rotor blade of a particular helicopter in forward flight, V = 260 km/h, at advance ratio $\mu = 0.33$ operate at angles of attack which vary periodically each revolution. On the advancing side, the angles of attack are small, while, on the retreating side they are large and, for the greater part, correspond to angles well above the static stall angle of a typical blade section. As a given blade advances, the tip vortex from the preceding blade may pass closely underneath it, as indicated in figure 25 for $\mu = 0.2$, and the point of intersection moves inward along the blade as the azimuthal angle ψ increases.

It is so possible, at the cruise forward speed of a helicopter requiring a total rotor thrust S for supporting its weight and air drag, to evaluate appropriate values of C_L, C_D, U_p and U_T as necessary for applying eq. (8).
For each azimuthal angle ψ (referred to a blade of the rotor) and for all the blade elements, and the b blades of the rotor, we may get the total torque Σ dQ required to overcome the induced drag and the profile drag. Adding to such a total torque the torque necessary to turn the tail rotor, the value of $(Q_{av} \pm \Delta Q)$ in eq. (2) is obtained. The periodic distribution of the rotor torque is shown on figure 26.

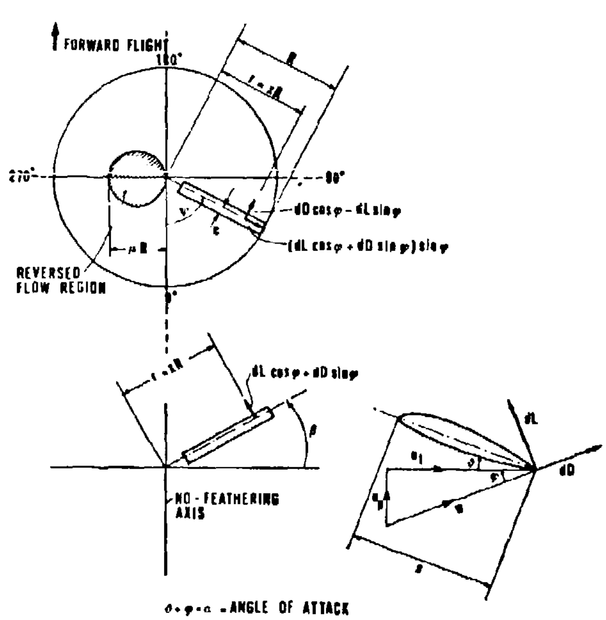


Fig. 23 - Variation of blade torque with azimuth angle in forward flight.

The average torque $Q_{d\ av}$ for b constant chord blades, due to profile drag, is

$$Q_{d\ av} = (b/2\pi) \int_0^R \int_0^{2\pi} \tfrac{1}{2} \rho\ U_T^2 \cdot C_D \cdot c \cdot r \cdot dr \cdot d\psi =$$

$$= (b/4\pi)\, \rho\, c\, \Omega^2 R^4 \int_0^1 \int_0^{2\pi} C_D (x + \mu \sin\psi)^2 \cdot x \cdot dx \cdot d\psi = \qquad (10)$$

$$= \rho \cdot b \cdot c \cdot \Omega^2 R^4 \cdot C_{D\ av} (1 + \mu^2)/8$$

assuming an average value $C_{D\ av}$ of the drag coefficient.

The average torque $Q_{i\ av}$ for b constant chord blades, due to rotor induced drag, is

$$Q_{i\ av} = -(b/2\pi) \int_0^R \int_0^{2\pi} \tfrac{1}{2} \rho\ U_T \cdot C_L \cdot U_p \cdot c \cdot r \cdot dr \cdot d\psi =$$

$$= -(a\ b\ \rho\ c\ R^2/4\pi) \int_0^1 \int_0^{2\pi} (U_T\ U_p\ \theta + U_p^2)\ x \cdot dx \cdot d\psi \qquad (11)$$

where it has been considered $C_L = a(\theta + \varphi)$, with a constant slope lift a.
To solve eq. (11), it is necessary to introduce the rotor thrust dS on each blade element as expressed by eq. (9), the force component dH perpendicular to the no-feathering axis, i.e.

$$dH = (dD \cos\varphi - dL \sin\varphi) \sin\psi - (dL \cos\varphi + dD \sin\varphi) \cdot \sin\beta \cdot \cos\psi \qquad (12)$$

and the air induced velocity profile, Ref. 5.

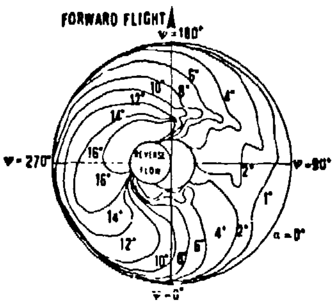


Fig. 24 - Angle of attack distribution with non uniform downwash for $\mu = 0{,}33$.

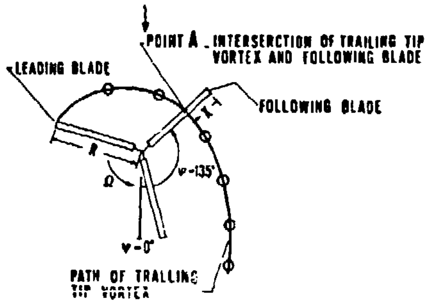


Fig. 25 - Intersection of following blade with tip vortex trailing from a leading blade for $\mu = 0.3$.

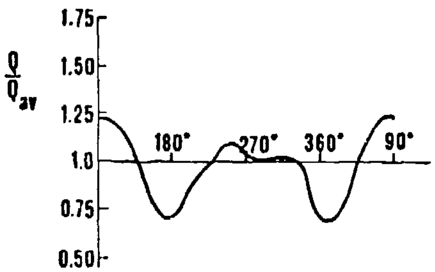


Fig. 26 - Variation of blade loading with azimuth angle.

The average torque Q_{av}, as in eq. (2), is obtained adding the required tail rotor torque to the values carried out with eqs. (10) and (11).

Finally, for each azimuthal angle ψ, the values of $\pm\Delta Q$ and $\pm\Delta Q \cdot \Omega/\omega$, as in eq. (2) and figure 26, and $\pm d\omega_e/dt$, as in eq. (4), are obtained, for the engine (and the transmissions) transient response.

Rotor torque in "over-speed" forward flight and in transient maneuvers - During "over-speed" forward flight and in transient maneuvers, vortex wakes spring from the individual blades and form a complicated downwash pattern, as the vortex elements spiral downwards below the rotor plane because of high Mach number and changing incidences. Moreover, effects of

flexibility may determine complex blade bendings and coincidence between the frequencies of the blade motion with the frequencies of forced motion or the natural frequencies of other parts of the helicopter. The periodically varying aerodynamic loads along the rotor blades transmit vibrations to the hub. Aeroelastic coupling of rotor blades determines flap-lag instability, pitch-lag instability, pitch-flap flutter, blade weaving, and stall flutter.

Large high-frequency oscillations in pressure, torsional moment, lift and aerodynamic moment on the retreating side, are detected during a maneuver. This response is the result of dynamic stall induced, figure 25, by the previously formed tip vortices, which, under the maneuver flight condition, pass under the blade at the azimuth positions over there indicated.

Rotor aerodynamics simulation programs are under development for a broad range of high speed and transmission flight conditions. From them, it will be possible to perform detailed analysis of rotor torque variation, more critical than that on figure 26. Such aeroelastic rotor analysis contains the following items:

- accurate representation of the rotor dynamics, including the effects of the centrifugal force field, blade twist, mass distribution, stiffness distribution, and the coupling effects between in-plane, out-of plane, and torsion displacement of the blades;
- the blade element aerodynamic coefficients C_L and C_D for stalled and unstalled conditions;
- the helicopter's response to pilot control inputs, gust and other externally applied forces.

However, prediction of the rotor torque Q azimuthal signature for "over-speed" forward flight and transient maneuvers is a complex task involving a high degree of idealization and many assumptions. When the rotor system concerned is hingeless, or semi-rigid, the complexity of this task is amplified by a strengthening of the inter-relationships that exist between the three fundamental loading actions of overall aircraft trim balance, rotor system oscillatory loading and vibratory forcing of the airframe.

From the other hand, the rotor torque, Q and Q_e, azimuthal signatures, as in eqs. (1), (2) and (4), may be directly measured in flight. The collected data can be used to analyze the engine component behaviour and the aerodynamic load distribution in "over-speed" forward flight and transient maneuvers.

According test flights, limitations are established for each type of helicopter. For instance, in any kind of maneuver, the flight speed must never overcome the maximum forward speed in the same condition.

Quite high inertial loads derive, in military helicopters, from maneuvers like the ones shown on figure 3, 6, 7, 8 and 9. With load factors as high as 3.5 g, sometimes istantaneously higher, serious mechanical damages and failures on engines may be involved. With negative load factors, as the ones on figure 8, the distorted inlet flow may cause induced engine instability.

ENGINE RESPONSE TO DISTORTED INFLOW

Flight in turbulence and abrupt maneuvers might induce distorted inlet flows and engine performance degradations. The combination of these phenomena to high amplitude (and/or frequency) rotor torque fluctuations may have serious consequence on the engine operation.

The collapse of axialsimmetry of the flow in the compressor, at rotational speed lower than the nominal speed, causes rotating stall. Although this phenomenon is confined only to a part of the compressor at a given moment, as in a abrupt maneuver in air turbulence, and does not severely affect the performance map of the compressor, it is more dangerous than other unsteady phenomena to the survival of the compressor. In fact, the rotating stall is comprised of a wide range of exciting frequencies which can include self frequency of one of the blade rows. The rotating stall usually occurs close to the compressor surge line, where the blade boundary layers are thick. In this region, a wall stall or surge configuration usually appears before the rotating stall can stabilize.
Blade stalls and rotor-induced surges are then occurring when the distortion intensity is such that steady or surge pressure ratio limits are reached.
During engine surge, dynamics loads stress fixed and moving blades and components.
Compression and expansion wave over and under-pressure relative to pre-surge steady levels occur as a consequence of flow breakdown within the compressor of the gas producer.
Possible indirect effect are the beam-like vibrations of the shaft, and the flexural modes of forced vibrations of the discs. These disc modes, or blade and disc modes of blade disc assemblies, consist of alternating patterns of axial displacements with increasing numbers of diametral modal lines and circular modal lines for higher natural frequencies. Failure prediction can then be conducted by comparing with the results of rotating disc fatigue, testing under controlled conditions of energy input.
The transmission (from the main rotor to the engine) damping, is an important means of reducing blade amplitudes. The phenomenon implies the dissipation of energy and the reduction of wear of a critical part in the case of structural damping or the generation and propagation of cracks along the internal grain boundaries of metallic alloys. The axial machines have to be designed to withstand more severe vibration environments by using composite materials.

Engine new technology is developing dynamic components (which have greater efficiency, reliability and ease of maintenance), advanced materials (metals and composites), and advanced lubrication concepts. High dynamic rotor loads in turbulence limit the high speed and maneuverability capabilities of helicopters. Dynamic loads influence the reliability and maintainability characteristics of a helicopter and,

hence, its life-cycle cost. The goal of the technology program is to predict engine dynamic performance capabilities during high speed forward flight and transient maneuvers.

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HELICOPTER PROPULSIONS SYSTEM

1. VIBRATION PREVENTION SYSTEMS ON HELICOPTERS
2. PROBLEM OF NOISE IN THE CABIN

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SUMMARY

This paper presents two different facets of the propulsion system/structure interaction.

The first interaction covered in chapter 1 deals with low-frequency vibration problems associated with the operation of the main rotor.

These problems are either mechanical-material fatigue ans stressing-or related to comfort. Chapter 1 describes the means implemented and the research done at Aerospatiale to reduce excitations and filter their transfer from the source to the cabin.

The second interaction type presented under Chapter 2 deals with cabin noise (frequency between 100 Hz and 10 kHz) associated with the operation of the reduction gear box. The vibratory energy generated at the gears propagates all the way bo the passengers via the air and the structure.

Chapter 2 describes the research, technological means and methods used at Aerospatiale to cut the intensity of transmission noise sources, the energy transfer between such sources and the structure and the energy radiation from the structure to the passengers.

CHAPTER 1 - VIBRATION PREVENTION SYSTEMS ON HELICOPTERS

1. INTRODUCTION

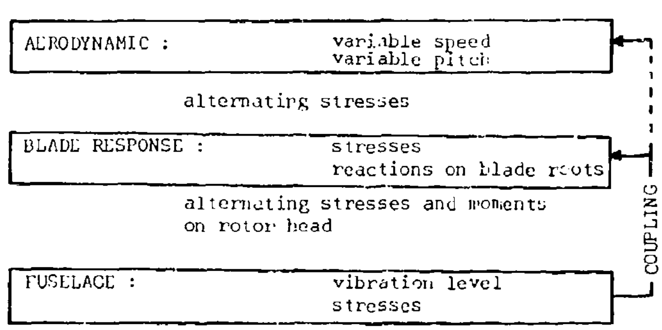
Vibration problems occur in a very special form on helicopters ; the rotor is a powerful vibration generator, causing problems that are specific to this type of machine, and one of the most acute is the problem of forced vibration.

This causes:

(a) alternating stresses throughout the aircraft, leading to problems of fatigue.

(b) cabin vibrations, which are a major factor in an aircraft comfort rating.

The diagram below summarizes the development of forced vibrations in the whole aircraft.



These alternating stresses are periodic, and their fundamental frequency is the rotational frequency of the rotor.

Depending on the blade characteristics, these stresses are amplified or reduced causing stresses and reactions in the rotor head. It is these reactions which cause the stresses and moments that make the fuselage vibrate.

The resulting fuselage response depends on its dynamic characteristics and on the filtering devices fitted between the rotor and the fuselage.

It is therefore important to select the dynamic characteristics of the blades and fuselage in such a way that their response to aerodynamic stresses will be minimal.

Aérospatiale invest much effort in the design of blades and a fuselage that are dynamically optimized. To take the example of the 365N and 366G, three 'generations' of blades were built and tested during the process of optimization.

It should be mentioned that the fuselage reaction on the dynamic condition of the blades, and the effects of blade movement on the aerodynamic stresses, can make the problem of forced vibrations more complex.

The considerations of better performance, multiple roles, improved comfort, new echnology (new materials, etc...) normal variation in production, and uncertainty at the design stage as to the aerodynamic problems of forced vibrations, have led to the following requirements :

(a) either that the moment giving rise to vibration should be minimized constantly, no matter what the flight configuration and dynamic signature of the aircraft. These methods are known as 'active control'.

(b) or that the dynamic stresses transmitted by the rotor to the fuselage should be filtered. These are called 'filtering methods'.

This paper will describe the direction taken in the use of these techniques by Aerospatiale in Marignane.

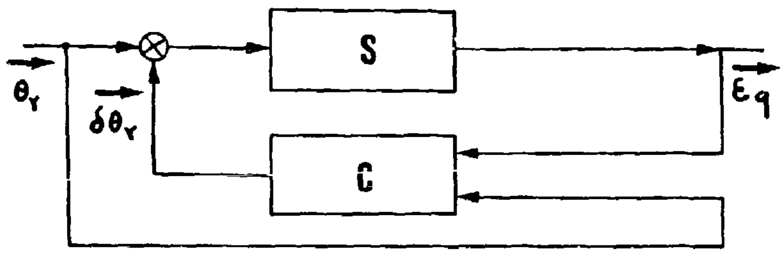
2. ACTIVE CONTROL

Purpose : to reduce vibration stresses by multicyclic variations achieved by swashplate or blade flap control. See fig. 1.

Method :

Vector of multicyclic law Fourier coefficients

Vector of vibration signals to be minimized



Role of C - harmonic analysis of signals
- monitoring vibration level
- identifying and optimizing control

- Our experimental evaluation of this type of system leaves us every cause for optimism.

3. VIBRATION FILTERING

For a long time, Aerospatiale has been attempting to place filters between the rotor and the fuselage, to prevent transmission of the stresses causing vibration and reduce the resulting vibration levels.

SUSPENSION SYSTEMS

(a) The best-known example is the 'barbecue' which is a flexible element placed between the transmission assembly and the fuselage, to provide insulation.

This is illustrated in Figs. 2 and 3.

. The one-directional 'barbecue' on the SA 330 Puma.

. There have been many technological developments of this idea, leading to highly simplified and low-cost systems.

For example : The two-directional MGB suspension on the AS 350 Ecureuil/Astar. Fig. 4.

COUNTER-VIBRATION DEVICES

(b) The second method has been to use energy dissipators or counter-vibration devices. These may be placed on the rotor :

as rotor head counter-vibrators.

Fig. 5 shows a bifilar pendulum fitted on an Ecureuil prototype. Fig.6 shows the production Ecureuil/Astar counter-vibration device, with uses weights and springs.

Or they may be placed on the fuselage :

Fig. 7 shows the soft-mounted 'battery' resonator on the AS 360 Dauphin.

SUSPENSION SYSTEMS WITH ENERGY DISSIPATORS

(c) A combination of filtering through elastic components and energy dissipators lead to a very compact built-in resonator system, as for example LE SARIB shown on fig. 8.

4. CONCLUSION

Thanks to all the features we have described, the users of Aerospatiale helicopters are able to fly in a high degree of comfort.

The methods for overcoming the problem of forced vibrations, we have been discussing, are being further investigated with a view to achieving still better results, with a lesser penalty in terms of weight and, hopefully, price. Further improvements become all more essential as the aircraft speed performance increases.

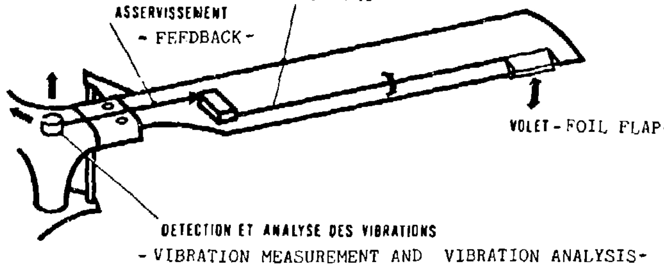


FIG 1

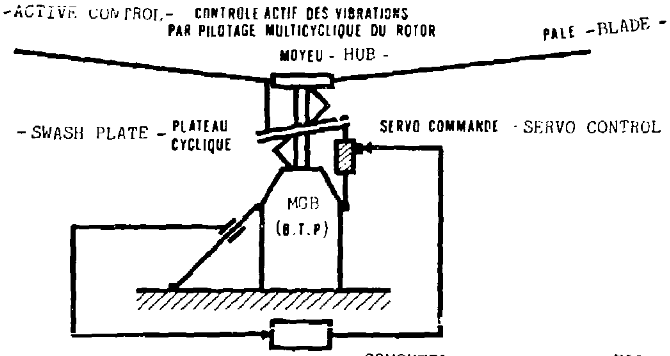
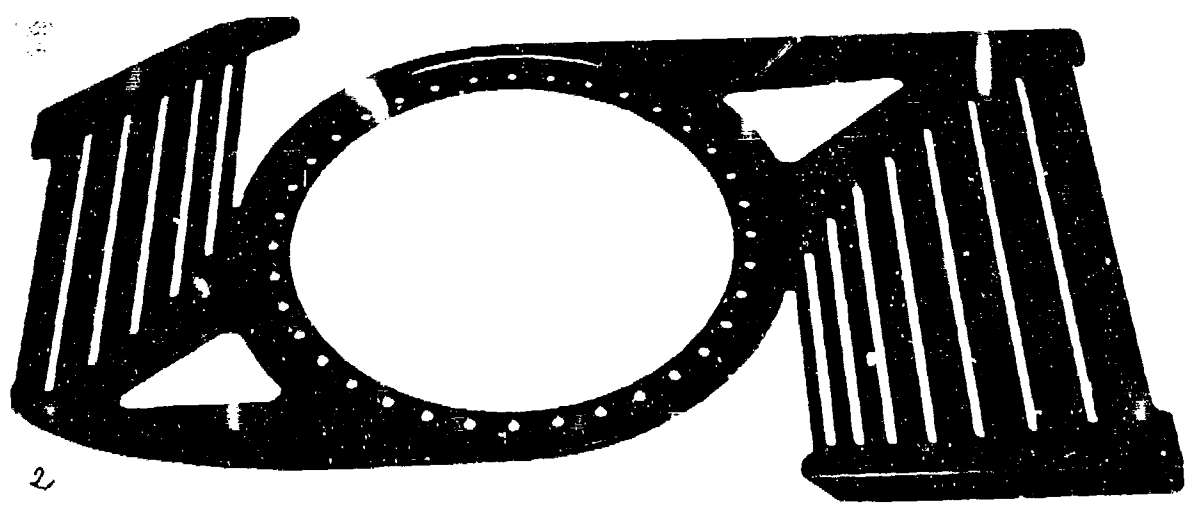


FIG 1



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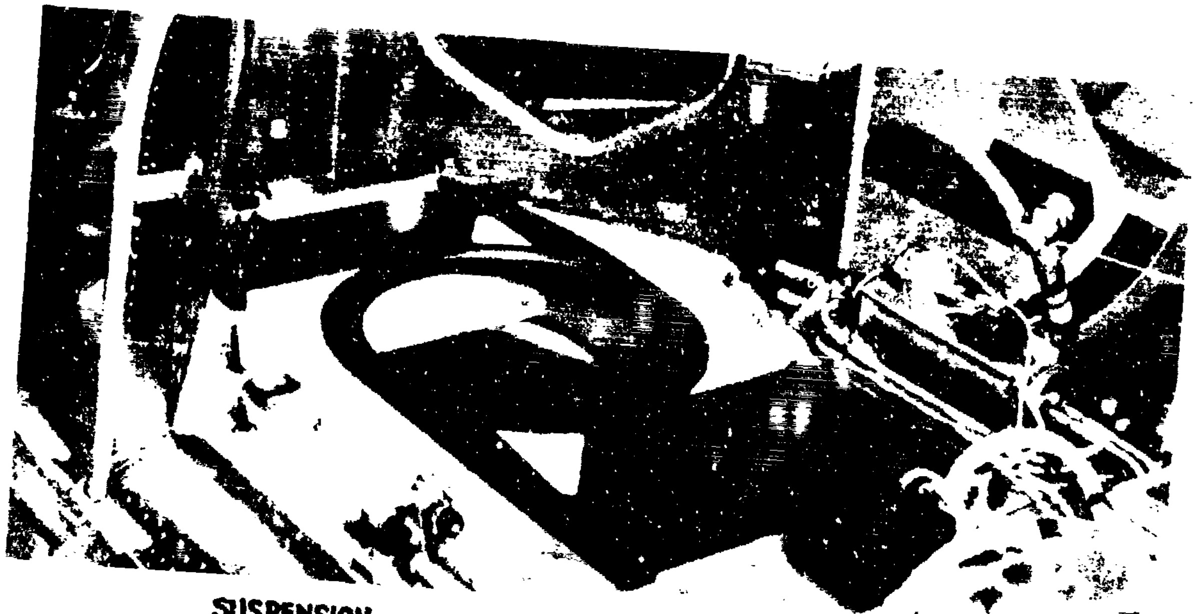
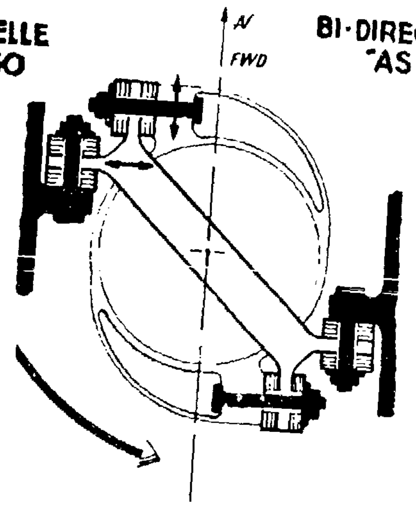
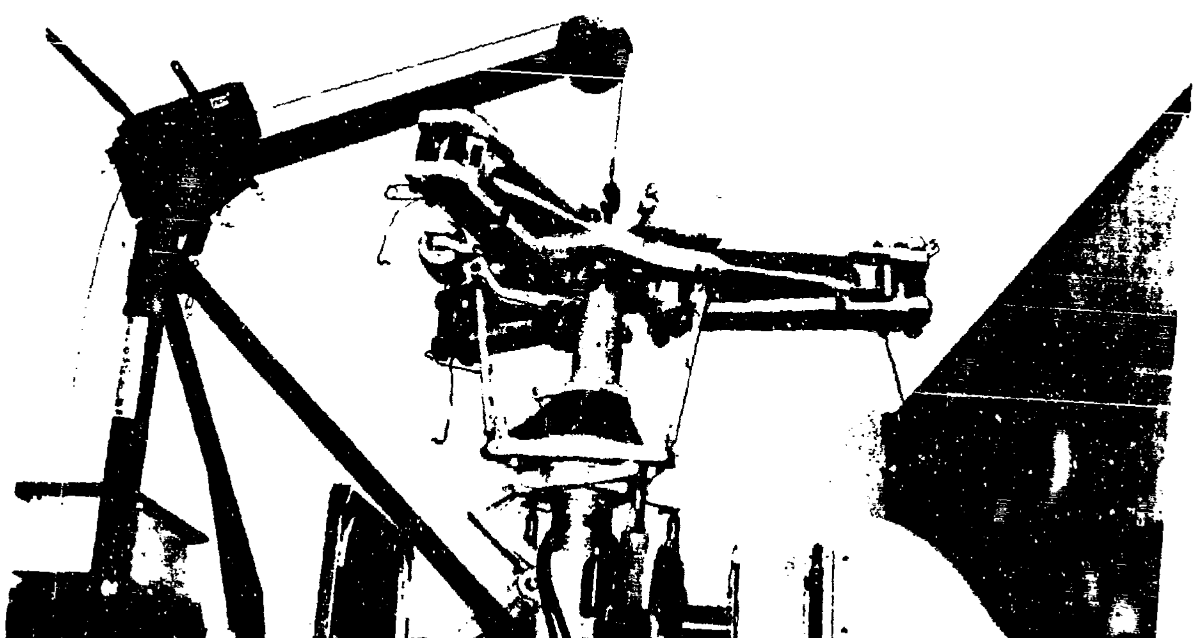


3

SUSPENSION
BI-DIRECTIONNELLE
TYPE A.S.350



BI-DIRECTIONAL MOUNT
"AS 350" TYPE



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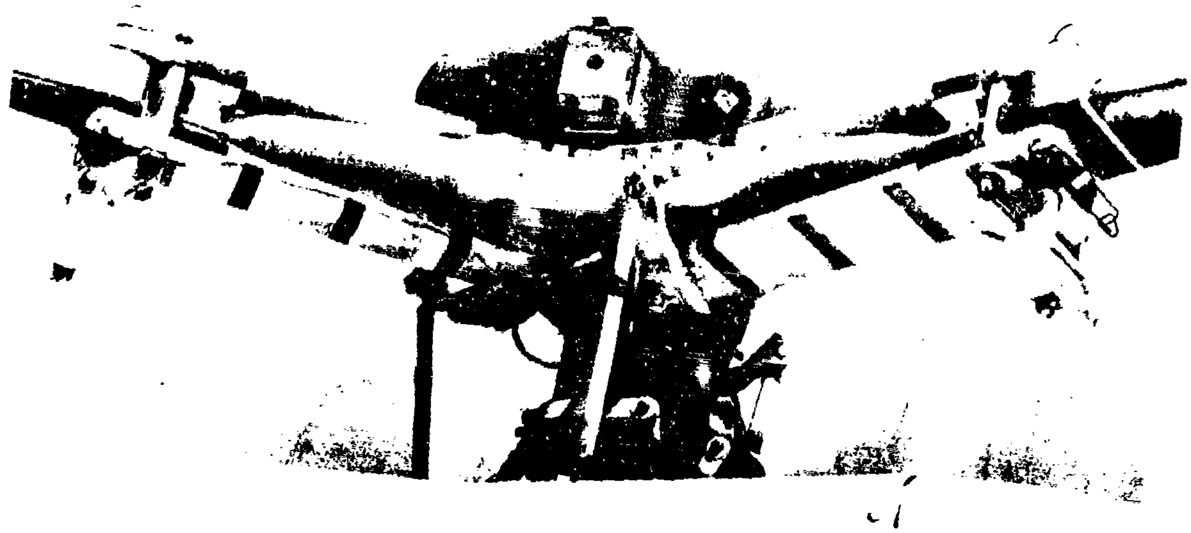
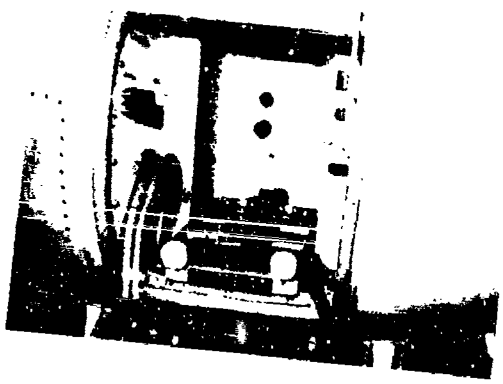




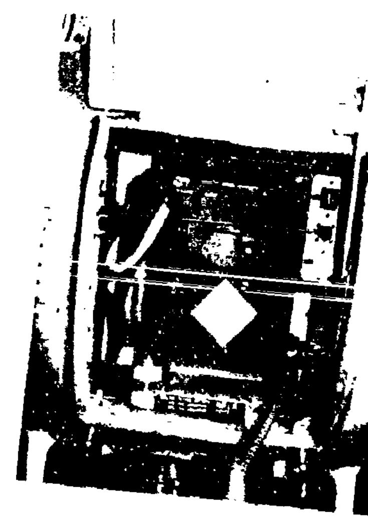


FIG 7

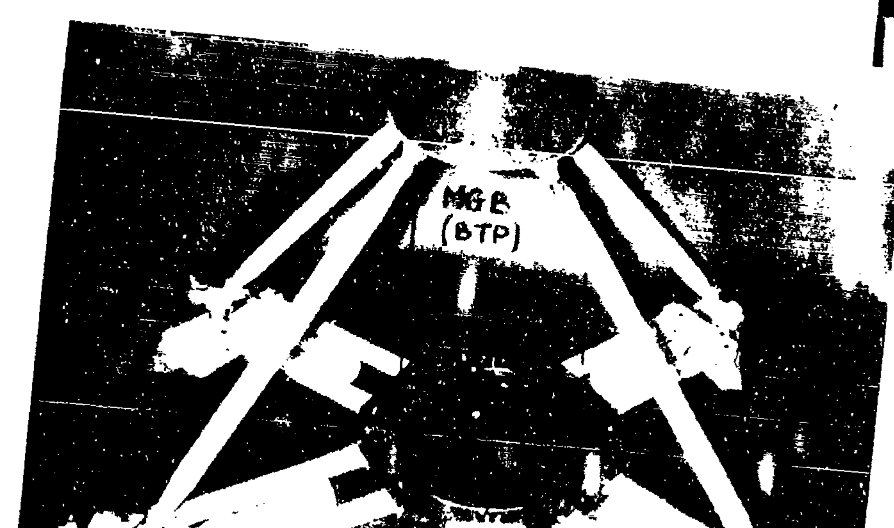


FIG 8

CHAPTER 2 - COUPLING BETWEEN THE HELICOPTER PROPULSION SYSTEM AND ITS STRUCTURE. PROBLEM OF NOISE IN THE CABIN

The total noise within a helicopter cabin is the sum of the rotor, engine and power train noises reaching the passengers via the air or the structure.

Figure 1 gives the result of a study that has led to the determination of the contribution of each individual source toward the total noise within a helicopter cabin. It can be noted that the noise induced by the reduction gearbox (MGB) is predominant in the range where frequencies are the most annoying (200 Hz to 10 KHz) and that the type of noise heard is above all structure-borne.

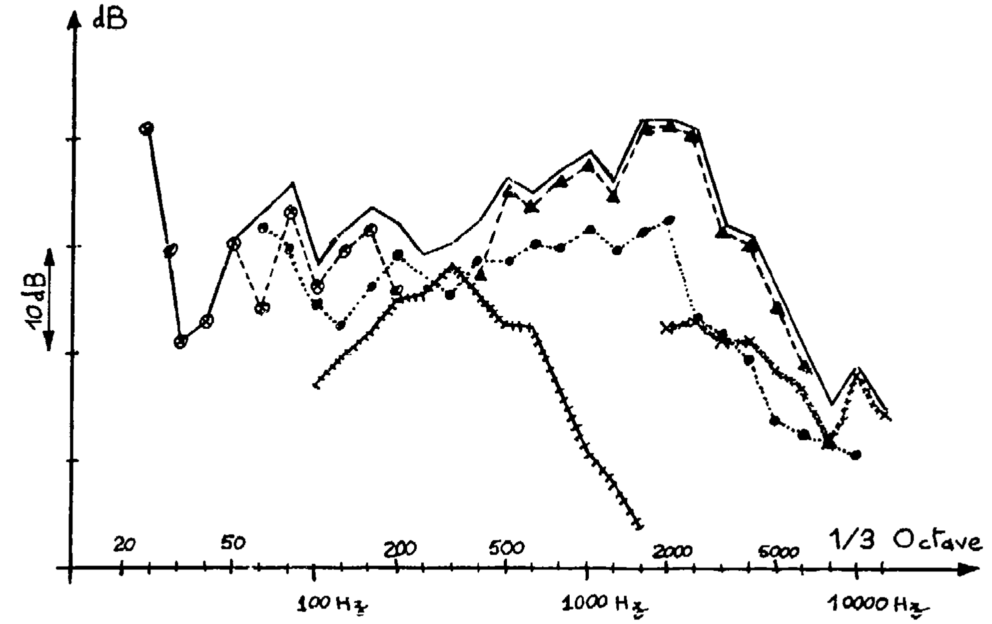


Fig. 1 - AS 365 Helicopter
Composition of SPL within cabin, at central aft location during flight at 135 kts

Stripped interior

————	Composite Sum
▲— — —▲	MGB Structureborne
●··········●	MGB Airborne
⊗- - - - -⊗	Main rotor Airborne
Xxxxxxxxxxx	Turbine Inlet - Airborne
-·-·-·-	Turbine Exhaust - Airborne
++++++++	Aerodynamic Excitation

Such high noise levels can be attributed to the fact that the systems for the transmission of power between the engines and the main and anti-torque rotors induce very substantial high-frequency vibrations.

Contrary to the case of airplanes where it is usually located far from the passengers, e.g. below the wings, the power train is attached directly on the cabin walls of helicopters. Thus a large part of this vibratory energy is introduced into the helicopter cabin structure which converts it in the immediate proximity to the passengers' heads into sound energy.

It must be noted that the level of noise inside the cabin of old-design helicopters is such that repeated exposure to such noise without any ear-protection may lead to an irreversible loss of hearing. Increasing the structure weight is the most obvious remedy to this noise problem.

However, as one knows, great efforts were made to improve the payload/empty weight ratio on helicopters, so we felt it necessary to look for technical solutions with a lesser weight penalty.

That is why we have been looking for several years at the problem of MGB-induced noise inside a helicopter. To that end, we have studied the methods that lead to : (see figure 2)

- a reduction in the generation of vibration sources,
- a limited transfer of vibration energy from the vibration sources to the structure, the main objectives being :
  1. to prevent the excitation frequency from matching the natural frequency
  2. to dissipate the energy before it reaches the structure
  3. to limit the vibratory-to-acoustic energy conversion process in the structure.

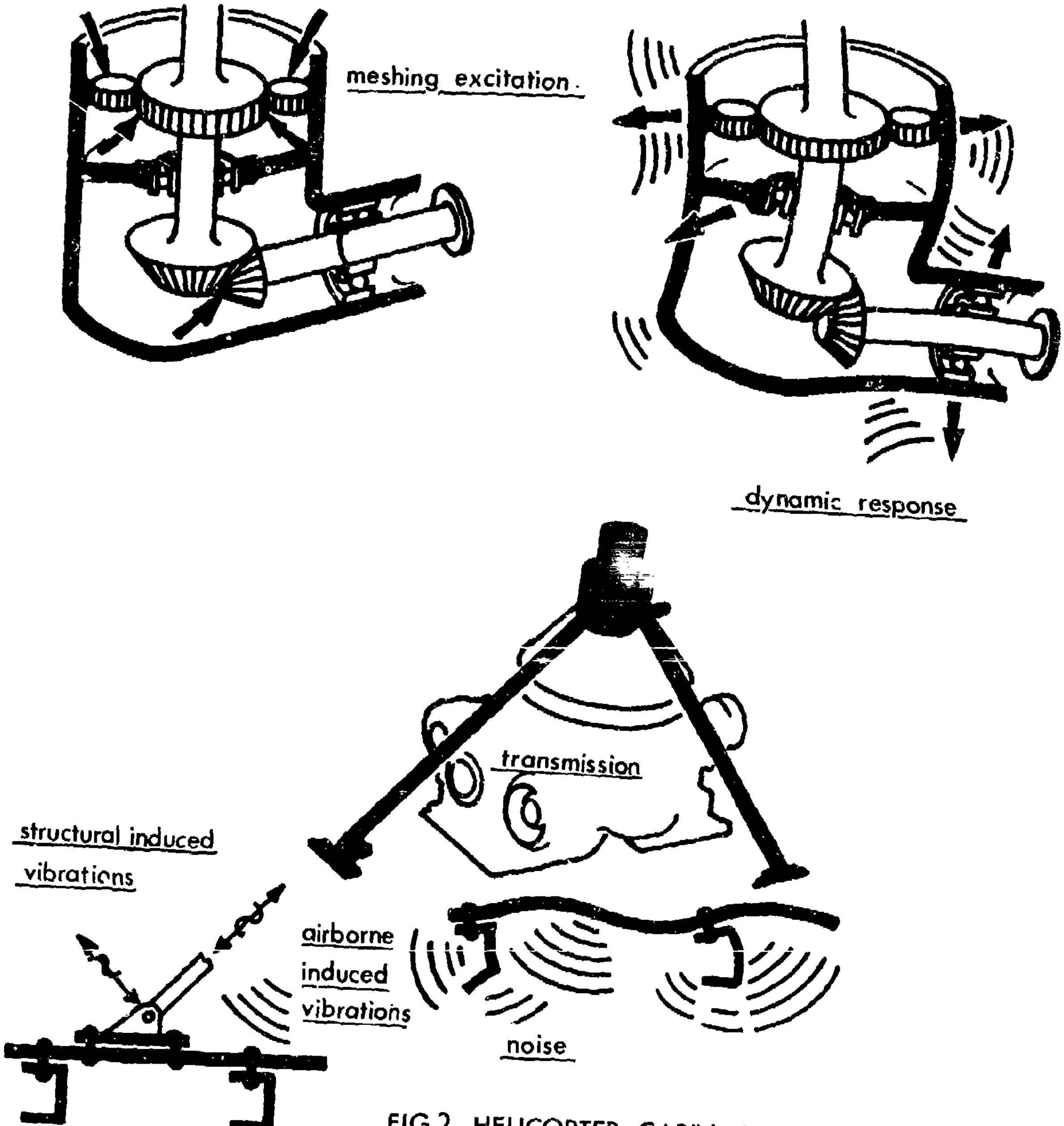


FIG 2 HELICOPTER CABIN NOISE GENERATION

1. NOISE SOURCE - MESHING NOISE REDUCTION

Gear meshing is a noise and vibration generator due to its design and realization ; angular meshing errors generate vibrations which will excite the structure (refer to figure 3).

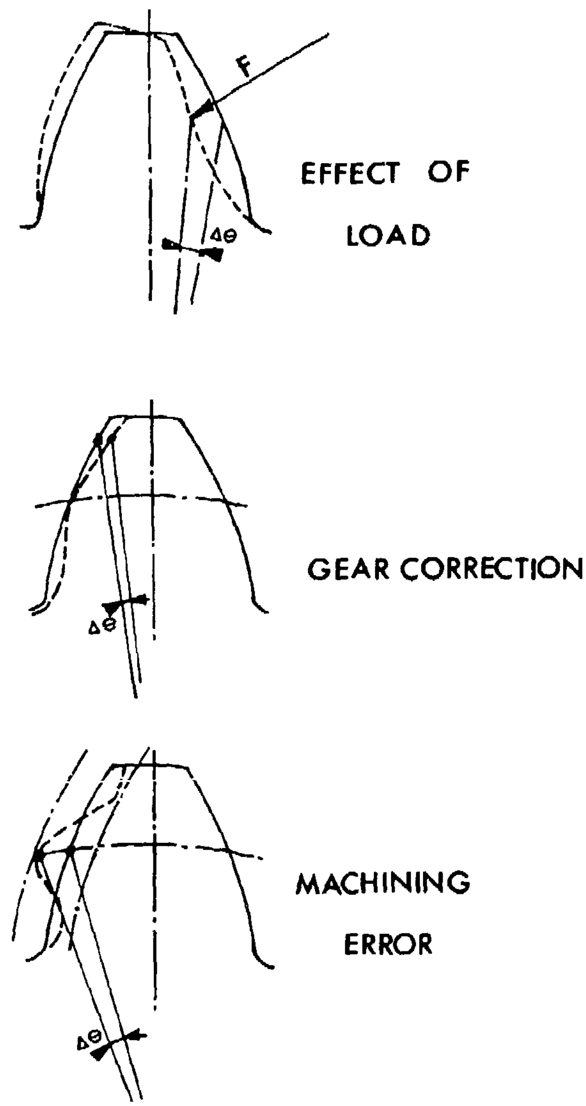


FIG.3-GEAR TOOTH DESIGN FACTORS IN THE GENERATION OF GEAR MESHING NOISE

Until the last few years, the compromise made, at the design stage, between the various gear parameters had as a main objective as ligth a weight as possible while ensuring a satisfactory service life.

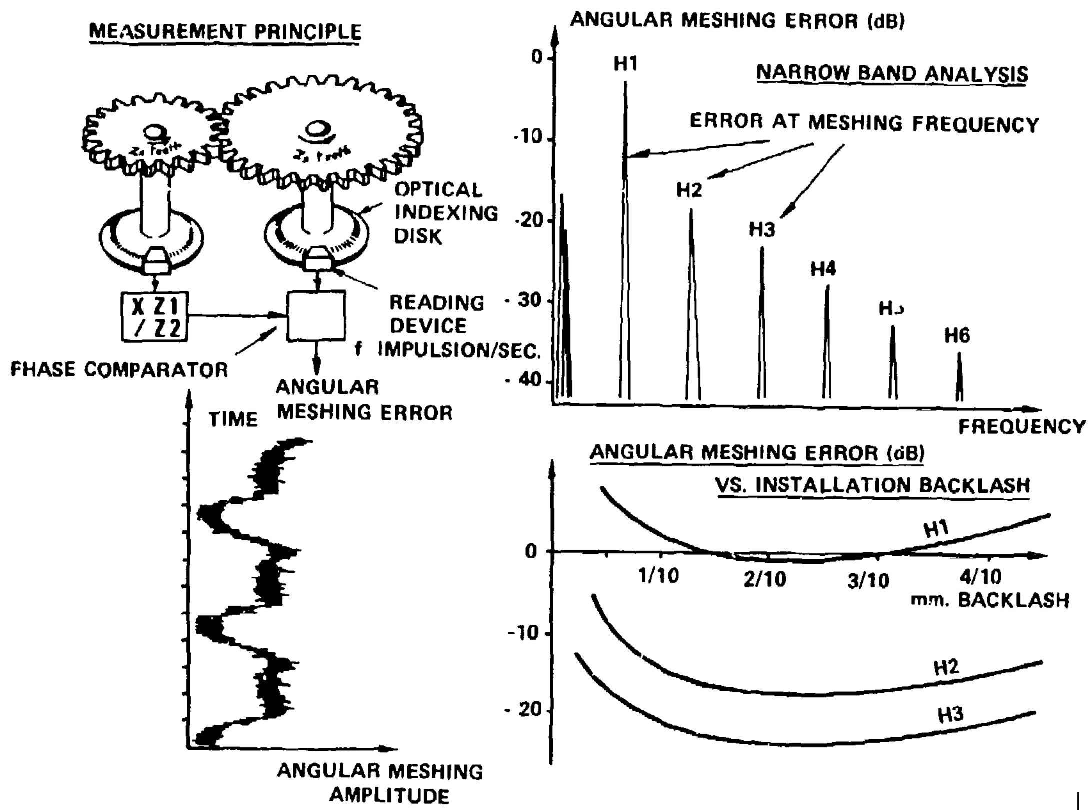
For that purpose, gear toothing was designed to work the closest possible to the maximum permissible stresses and specific pressures, but also to limit axial, radial and tangential loads on bearings.

This choice is the contrary of the continuous meshing concept, as the tooth bending increases with the load ; the low driving and overlap ratios achieved with low spiral angles or modules induce sudden load variations during meshing and do not ensure the compensation of machining errors which would require the simultaneous meshing of several teeth.

In a first stage, we have measured the angular meshing error on a pair of pinions, under no load, using a GOULDER MIKRON checking machine (refer to figure 4).

The results, in analog form, recorded on paper, clearly show the existence of tooth profile errors superimposed on an offset error or distortion of the basic circle. The spectral analysis of these analog signals allows the separation of these phenomena and the quantification of the effect of additional parameters, such as backlash.

- FIG.4 - ANGULAR MESHING ERROR -



Example of toothing geometry modification (see figure 5)

On the SA 365 and SA 360 main gear box, the input spiral bevel gear toothing has been re-designed taking the acoustic aspect into account ; the tooth bearing pattern has been optimized to ensure a better meshing continuity.

The gain achieved over the original meshing is approximately 15 dB and 12 dB.

- FIG.5 - EFFECT OF PINION DEFINITION CHARACTERISTICS ON GEARBOX MESHING FREQUENCY NOISE

	SA 360 MAIN SPIRAL BEVEL GEARS		SA 365 INPUT SPIRAL BEVEL GEARS	
	FIRST DESIGN	MODIFIED DESIGN	FIRST DESIGN	MODIFIED DESIGN
SPIRAL ANGLE	22° 30	35°	22° 30	40°
PRESSURE ANGLE	22° 30	20°	22° 30	20°
"MODULE" (Eq. to D.P)	5.434 mm	5.555 mm	5.14 mm	5.27 mm
DRIVING RATIO	1.221	1.254	1.275	1.234
OVERLAP RATIO	1.184	2.165	1.05	2.193
OVERALL RATIO	1.701	2.502	1.652	2.517
FUNDAMENTAL MESHING FREQUENCY NOISE REDUCTION	12 dB at 1850 Hz		15 dB at 2700 Hz	

Effect of tooth surface treatment on gear box noise

Introducing specific additives into the lubricating oil of the engine gear box allows a substantial noise reduction of 6.5 dB in 1/3 octave meshing frequencies thanks to the improved contact quality of gear toothing surfaces (see figure 6).

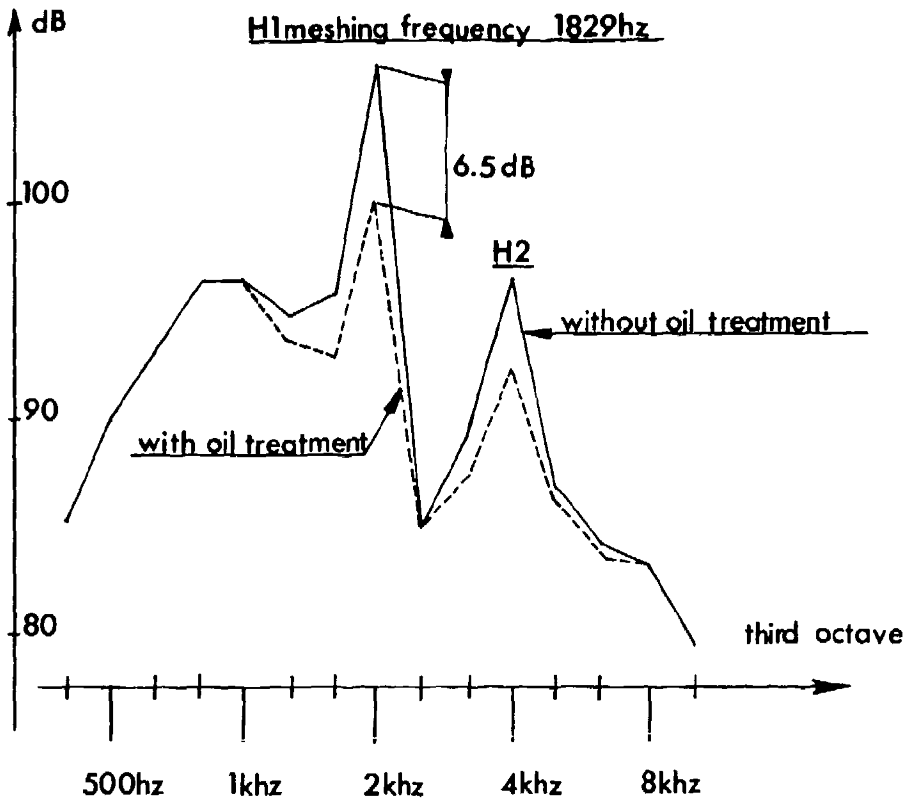


FIG.6. Effect of tooth surface treatment on gear box noise

2. TRANSMISSION BETWEEN SOURCE AND CABIN

2.1. Dynamic behavior of detail parts

If a meshing concept, taking the acoustic aspect into account, is a necessary condition to achieve a low noise level, it is not sufficient. In fact, in the transfer of vibration energy to the structure, the dynamic behavior of each of the components constituting the transfer path (pinions - shafts - bearings - casing - main gear box attachment fittings) has to be considered.

Axisymmetric part modes (pinions, shafts)

In a first stage, an experimental and theoretical mode determination has been made for the parts constituting the gear train.

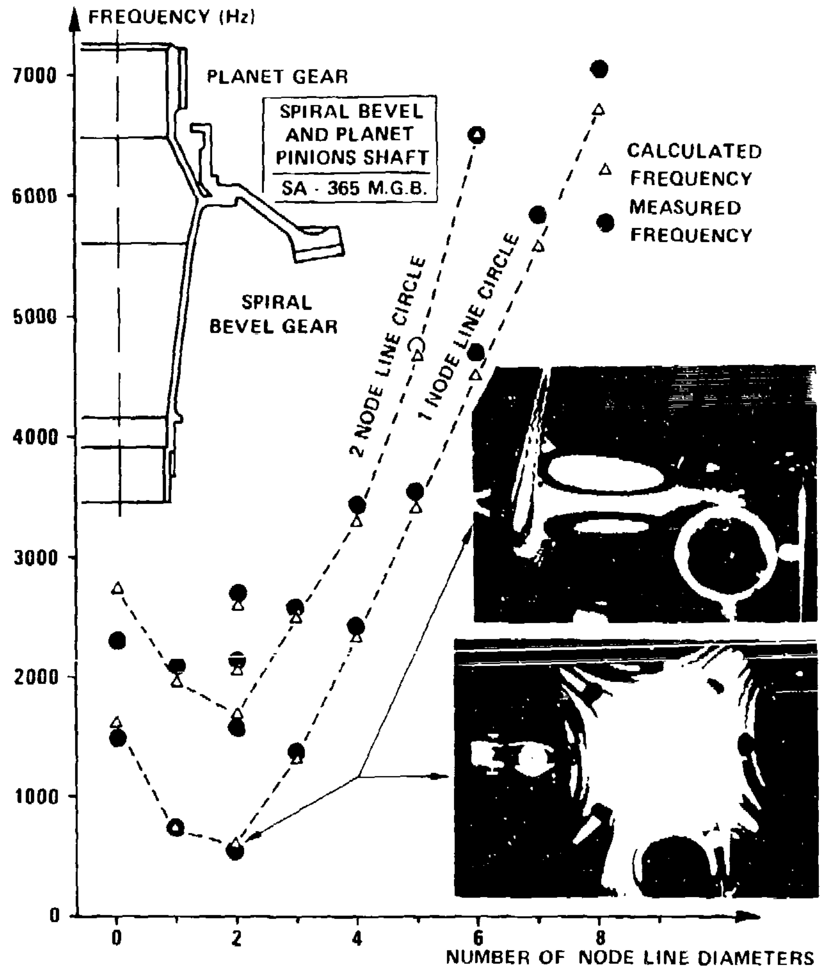
Refer to figure 7 for the results of a mode determination made, using a laser holography method and a finite element mathematical model, on a spiral bevel and planet gear assembly.

The mathematical model established allows the determination of the axisymmetric part modes under load and in rotation.

The agreement between modes calculated and those measured in laboratory using laser holography is excellent up to 7-8 KHz.

FIG.7. COMPONENTS DYNAMIC CHARACTERISTICS

COMPARISON BETWEEN THEORY AND EXPERIMENTS



The search for agreement between the main gear box natural and excitation frequencies (refer to figure 8) shows that it is difficult to design a complete main gear box in which no component natural frequency would be in concordance with a meshing frequency.

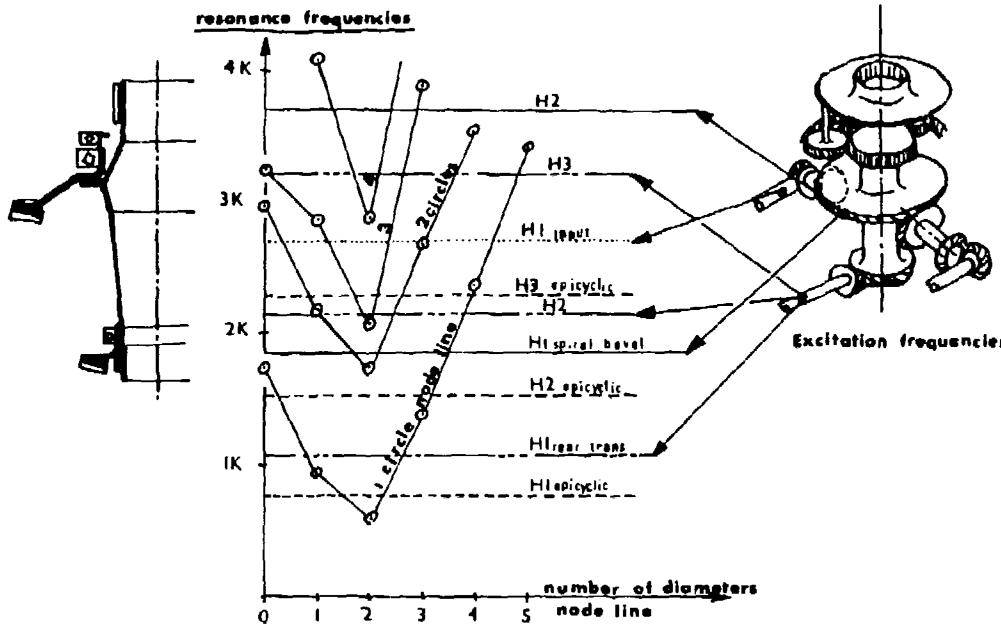


FIG.8 - Comparison of component natural frequencies with gear meshing frequencies.

This difficulty of mastering the full gear train dynamic behaviour has been checked on an actual main gear box in which the spiral bevel ring gear rigidity had been modified.

Figure 9 shows the changes in noise levels, measured on the acceptance test bench, for one of the spiral bevel gear meshing frequencies versus rotational speed and in two different configurations : initial ring gear and reinforced ring gear. According to the rotational speed, the modification may be beneficial or not, and for non negligible gains achieved at nominal r.p.m. at this frequency, there were appreciable losses at other meshing frequencies.

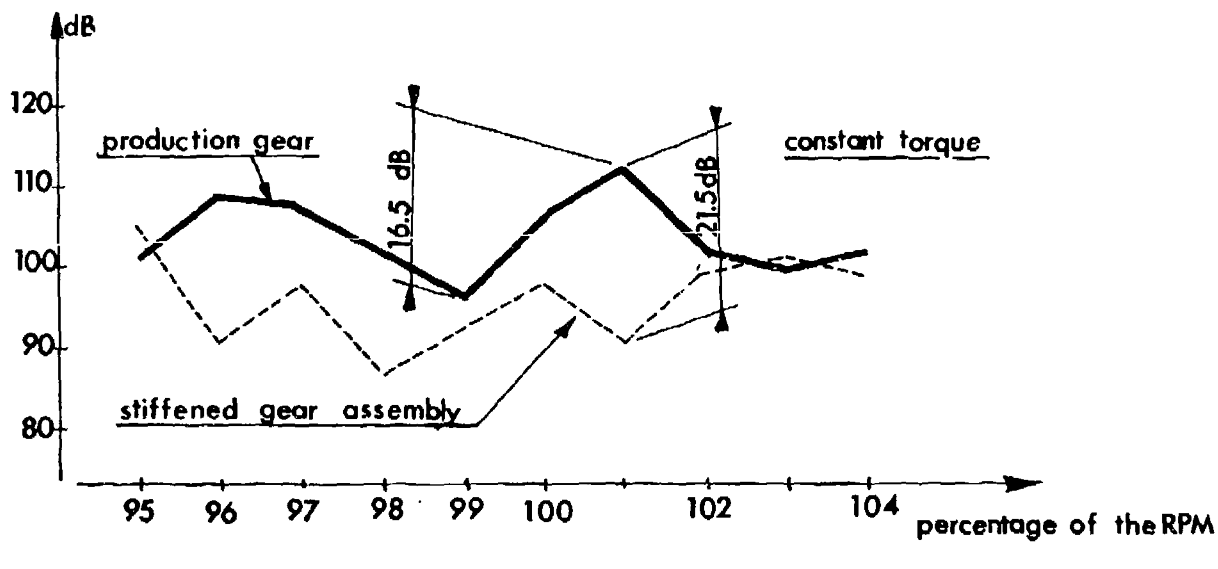
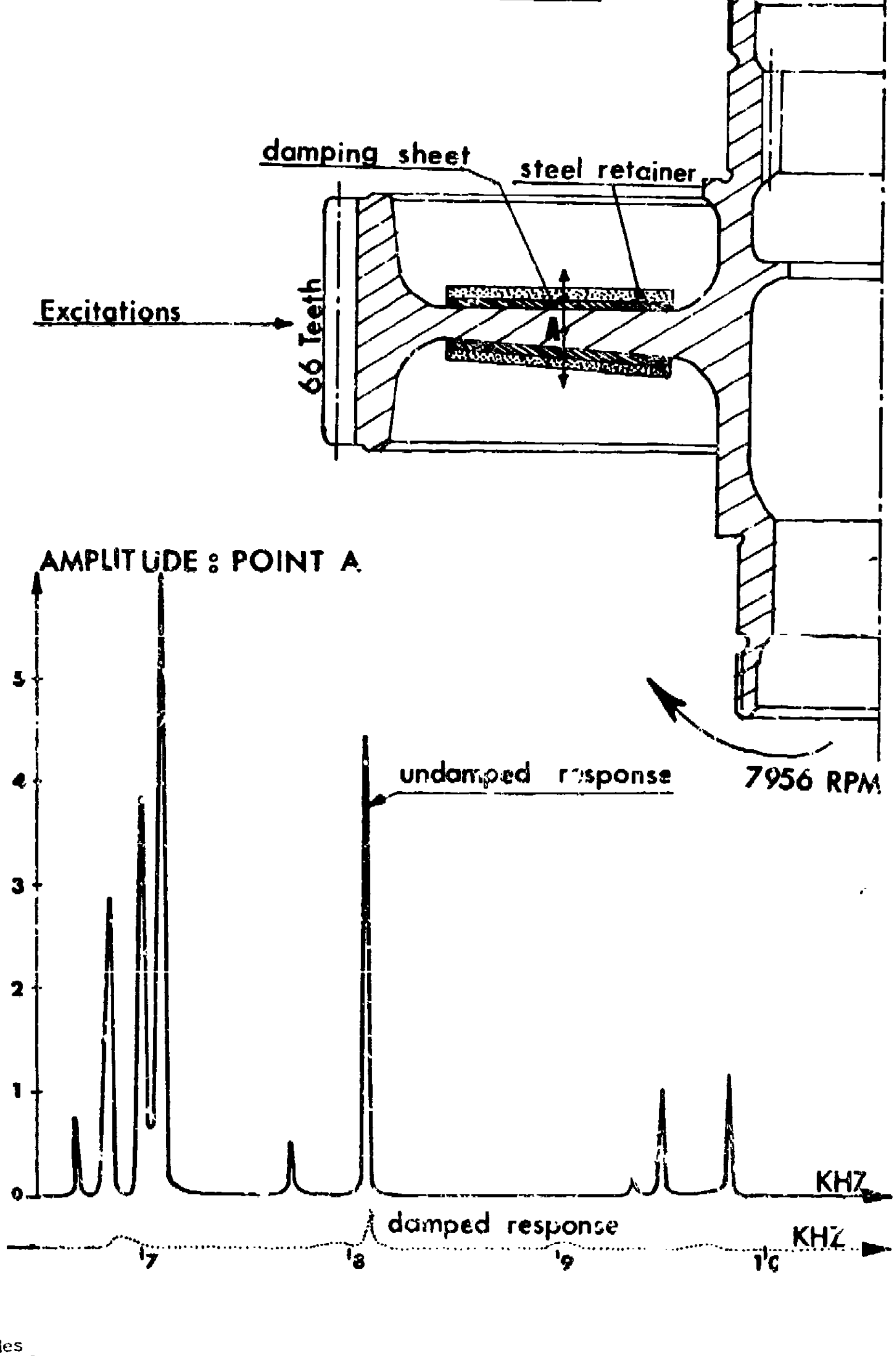


FIG. 9 - Influence of stiffening of spiral bevel ring gear on the gear noise (2nd meshing harmonic)

The introduction of some damping throughout the gear train seems to be a useful line to prospect with a view to reducing the gear train dynamic responses. A mathematical model established allows the determination of the forced response for damped axisymmetric parts.

Figure 10 shows the treatment applied on the web of a cylindric gear, and the damped and undamped responses of this web under harmonic meshing load.

FIG: 10 Noise and stress damped gear



Casing modes

Knowing the main gear box casing dynamic behaviour is a very important factor ; in fact :

- Due to the vibration of its wall, the casing is a source of noise.
- The vibration energy generated at the source and transmitted to the casing through the bearings will reach the structure through the casing attachment points (main gear box suspension bars and flexible mounting plate).

- The casing supports the shafts and thus ensures proper positioning of meshing gears, hence the risk of coupling between the excitation and casing response.

Modal determination in laboratory :

As for axisymmetric parts, modal determination has been made, in laboratory, on a main gear box casing using the laser holography method.

Figure 11 shows two examples of mode determination on a casing. On the prototype casing, it has been noted that a natural frequency of 1792 Hz was close to the spiral bevel gear meshing frequency of 1850 Hz.

A structural change (stiffening of casing through a rib located at mid-height) has relocated the natural frequency from 1792 Hz to 1850 Hz and generated a new mode at 1729 Hz. As there is a slippage of natural frequency according to the load (from 1792 Hz to 1850 Hz) the modified casing should no longer have natural frequencies in concordance with the spiral bevel ring gear meshing frequency. In fact, a gain of some dB's has been noted during the bench testing of this modified casing.

Provisional determination of casing modes :

The 1st vibration modes of a casing can be identified accurately through a calculation by finite elements. However, since calculations are of great magnitude because of the complex shapes of the casings it is not possible today to access high ranking modes through calculations (lack of accuracy, too long calculation times).

Damping treatment on the casing:

As in the case of detail parts of the transmission system, damping the MGB casing seems to be an efficient means for reducing vibration and noise for a slight weight increase.

Two approaches are being investigated presently :

- either a damping coat is applied on the casing surface.

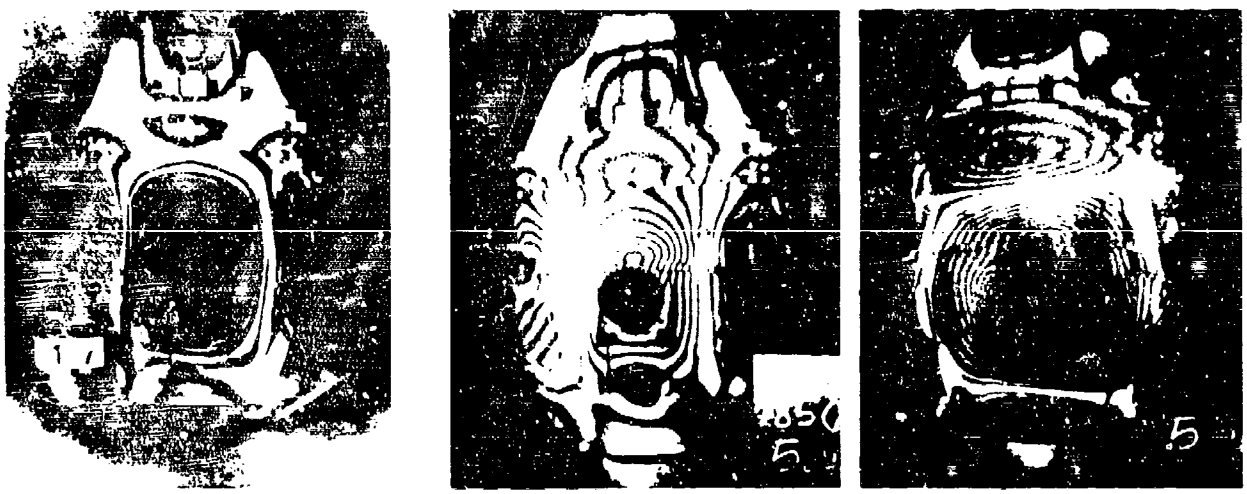
- or the casing is made of a high noise-absorption material.

A prototype coating has been tested on an MGB in flight and allowed a 3 dB cabin noise reduction for an addition of 1 kg extra weight.

- FIG. 11 -

MGB HOUSING DYNAMIC CHARACTERISTICS (SA 360)

MODE SHAPES BY HOLOGRAPHIC TECHNIQUES



NATURAL FREQUENCIES

1. PROTOTYPE GEARBOX — **FREQUENCY: 1792 Hz**

2. MODIFIED GEARBOX — **FIRST FREQUENCY: 1850 Hz** — **SECOND FREQUENCY : 1729 Hz**

2.3. Dynamic behaviour of complete main gearbox :

By controlling the dynamic behaviour of the simple components making up the MGB it is possible to avoid certain resonance problems. Yet the interaction of all these elements when the MGB is operating under load on a helicopter is not taken into account.

Two types of experiments were run on MGB's with a view to illustrating the rotation and load effects.

Response of complete main gear box on test bench (examples SA 360 gearbox)

(a) Effect of r.p.m.:

Variations with the rotation speed : noise, vibration or stress levels associated with each meshing stage during operation on a test rig were measured (see example on figure 12). These investigations made it possible to locate certain dynamic and acoustic problems during operation but it was difficult to process the result in the case of MGB's with numerous meshing stages.

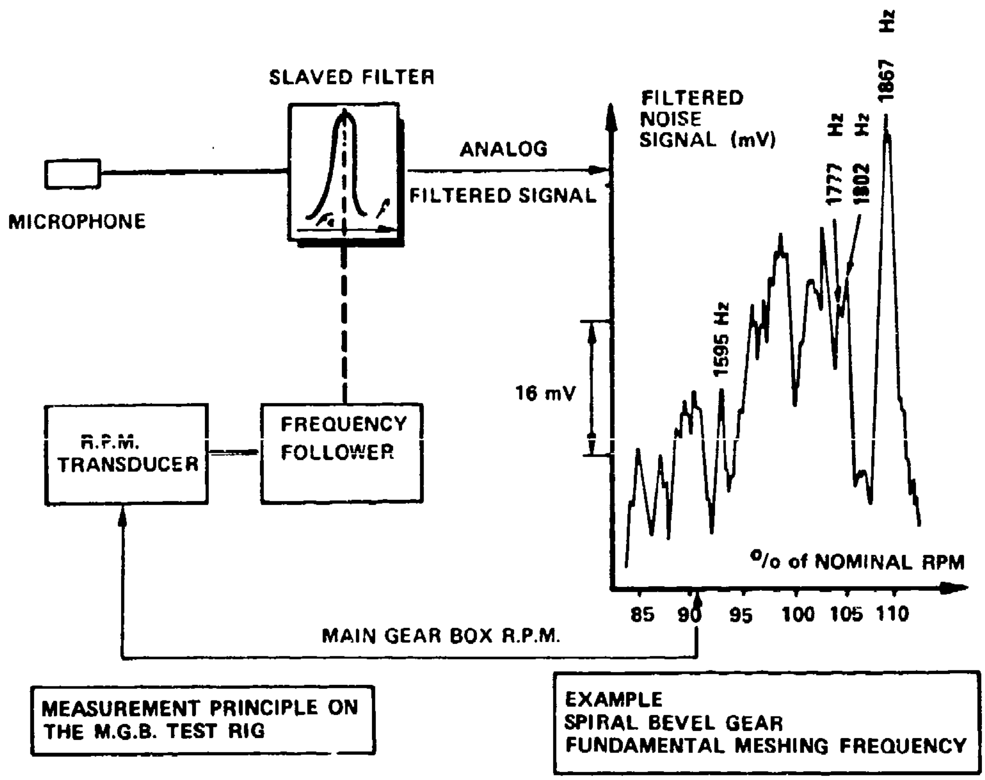
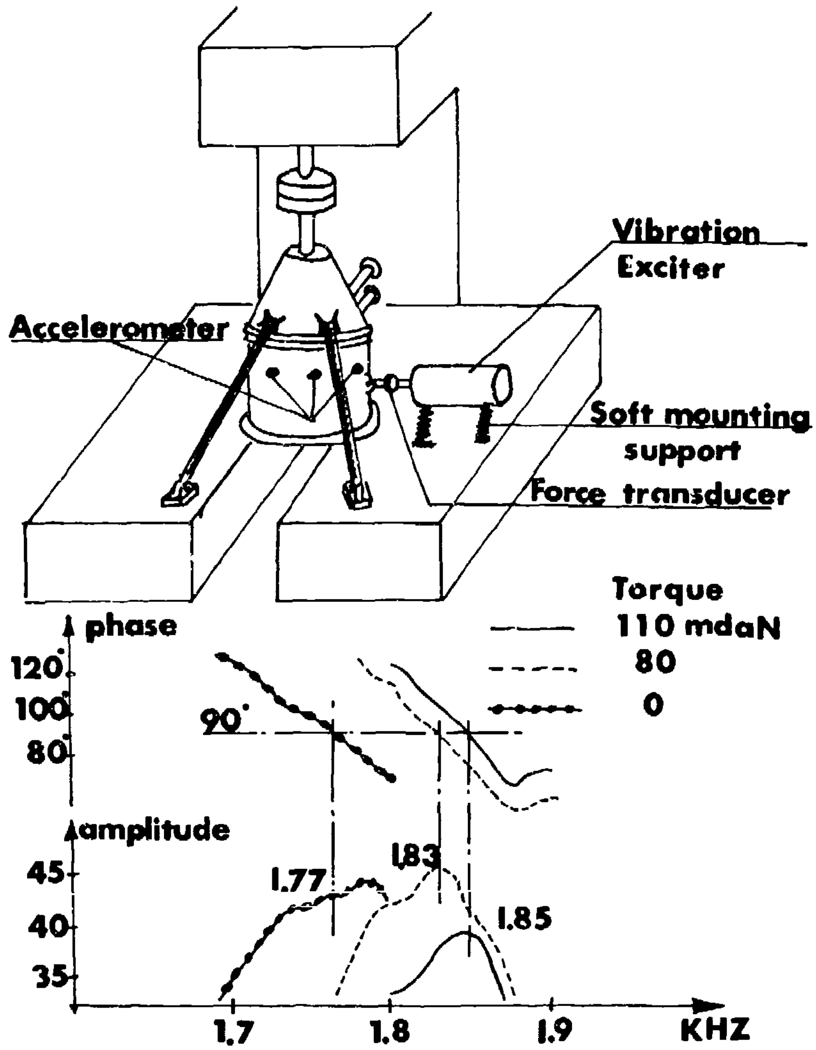


FIG.12 EFFECT OF R.P.M. ON GEAR BOX NOISE

(b) Effects of Torque :

To check the modes obtained in laboratory tests on a casing a bench accelerometric measurement (refer to figure 13 for set-up) showed there was really a very large response of the main gearbox casing at 1770 Hz and this frequency was moving towards 1850 Hz when torque was getting nearer the nominal load.



-FIG. 13- EFFECT OF TORQUE ON THE DYNAMIC BEHAVIOR OF MGB HOUSING

Other parameters like mounting clearances can introduce noise generating parasite dynamic behaviors .

We are presently focussing our efforts on the calculation of the dynamic behavior of a complete power train by taking into account all meaningful parameters.

This work turns out to be long and utterly complex and it will have to be continued for several years.

2.4. Main gear box suspension bar dynamic behaviour :

The main purpose of the main gear box suspension bars is to ens re the transfer of lift loads to the structure ; the attachments on structure and main gear box upper section are made through metal hinge fittings.

Therefore, the main gear box casing vibratory motions are transmitted to the structure without possibility of energy dissipation.

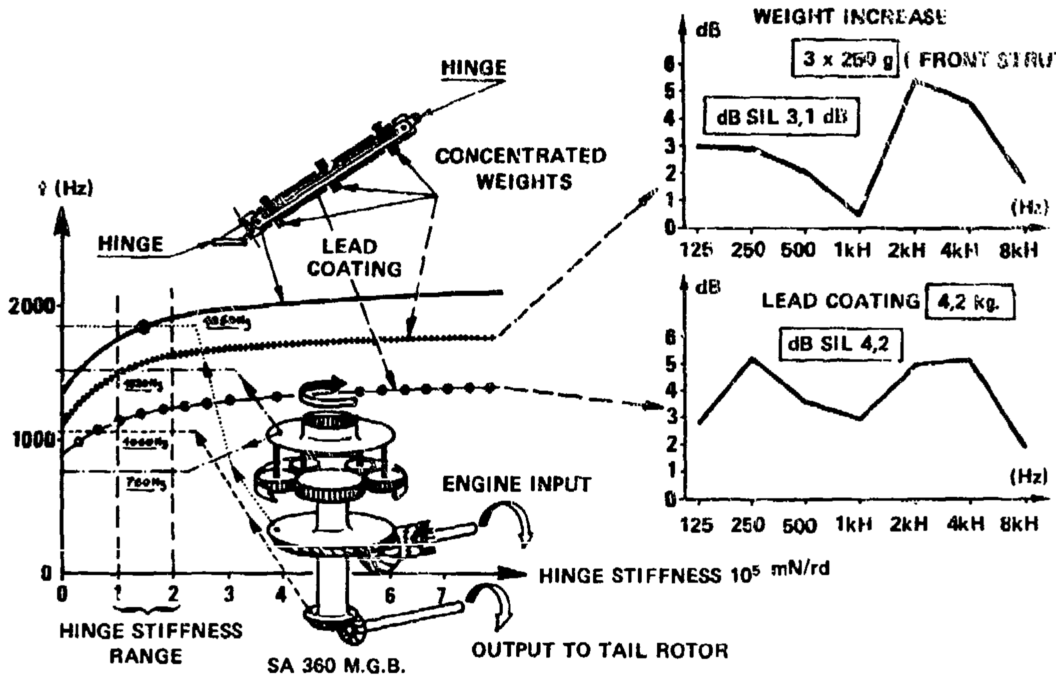
For the SA 360 main gear box bars, the first bending modes, in "free-free" configuration, have been determined in laboratory (excitation through B and K vibrating pot and accelerometric recording).

This mode determination, in laboratory, has allowed the validation of the mathematical model used to calculate the bending modes and the study of the effect, on the bars, of the hinges and weight (concentrated or distributed weights).

Figure 14 shows the results of the calculations made on this main gear box bar.

We can see the correspondence between the third bending mode frequency (1850 Hz) and the spiral-bevel gear meshing frequency, together with the large displacement to the resonant frequencies according to the type of weights added to the bars.

The efficiency of these weights has been verified in flight as, with 1.3 kilograms of lead distributed on the four bars, the mean noise level dropped by 4.2 dB SIL.



(1) THIRD BENDING MODE NATURAL FREQUENCIES (COMPUTED)

(2) IN-FLIGHT INTERNAL NOISE REDUCTIONS THROUGH TREATMENT OF M.G.B. STRUTS

FIG14 . EFFECTS OF MGB STRUTS DYNAMIC CHARACTERISTICS ON INTERNAL NOISE

Other alternatives making use of the flexibility effect of attachment points by interposing metal/rubber laminated elements were also investigated.Unfortunately this type of alternative is not efficient unless the stiffness values can be reduced to levels that, however, are not compatible with the elongation levels that are permissible on helicopters in flight.

2.5. Dynamic behavior of the structure. Conversion of vibratory energy into noise :

The vibratory energy introduced into the helicopter structure via the MGB attachment points spreads to the entire structure and thereby creates a complex field of vibrations. The structure components (metal sheets, frames, stringers ...) convert part of that vibratory energy into acoustic energy.

The characteristics of the vibratory field and the intensity and directivity characteristics of the acoustic field thus radiated depend on :

- the excitation levels at a given frequency
- the geometric characteristics of the elements making up the structure, their assembly mode and the nature of the materials used.

We have studied the radiation of elementary structures on the basis of Aerospatiale experiments but also on the basis of a large number of publications dealing with this matter.

The structure of first-generation helicopter cabins was made of metal and comprised thin aluminium alloy panels stiffened by stringers and frames.

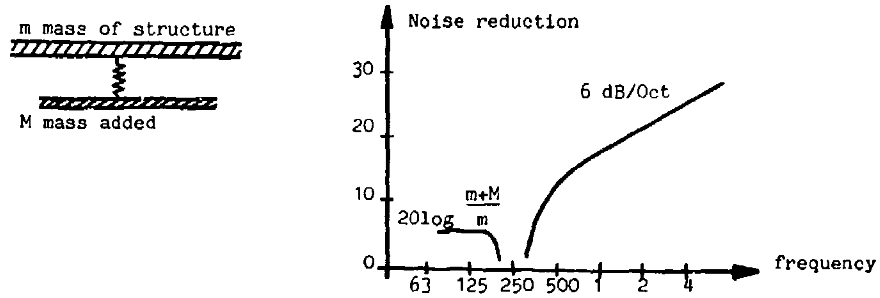
More modern helicopter designs make a wide use of composite materials and large size sandwich structures.

Both the dissipation of vibratory energy and the limitation of radiation on thin (metal or composite materials) panels with special treatments combining visco-elastic and weight characteristics are easy to devise. (see figure 15)

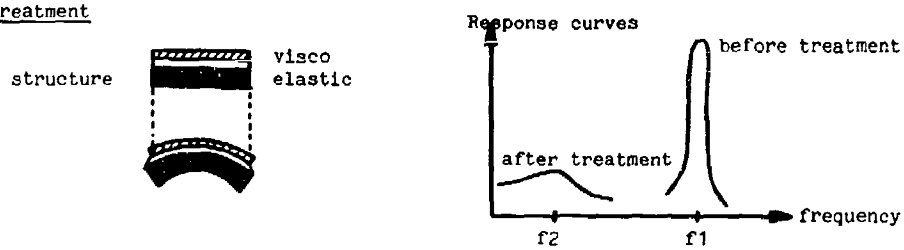
Usually visco-elastic materials subjected to shearing stresses are used for the sake of better efficiency.

-FIG.15- ACOUSTIC TREATMENT

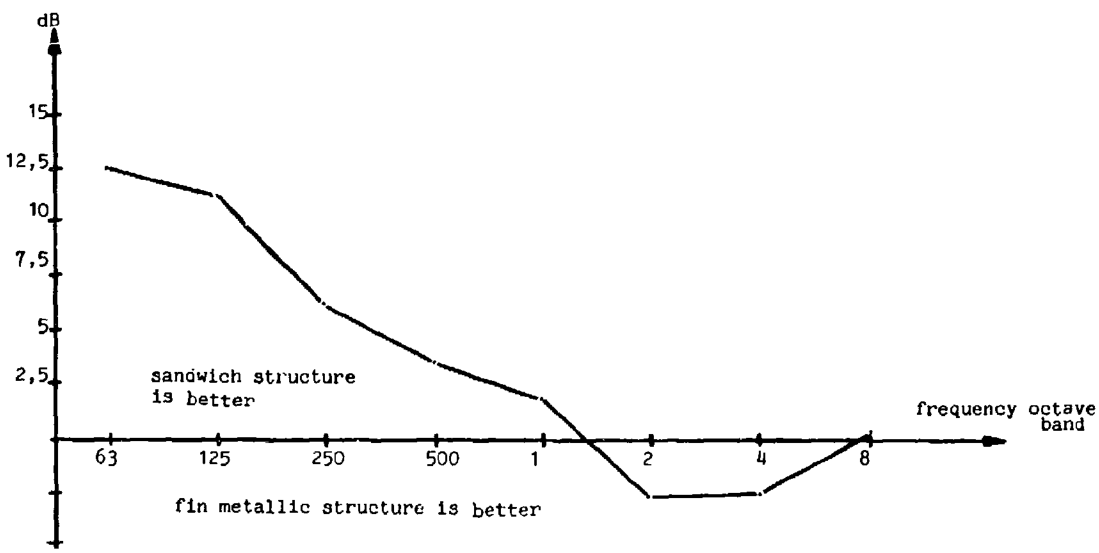
a) Mass effect



b) Damping treatment



-FIG.16- ACOUSTIC RADIATION DIFFERENCE BETWEEN A FIN METALLIC AND A THICK SANDWICH STRUCTURE



The main problem we face is the fact that the number of products available on the market today is very limited as they must be capable of meeting helicopter criteria : resistance to fire, high damping capability in a very wide temperature and frequency range, resistance to oils and fuels, long life etc...

Sandwich structures are more difficult to treat as their radiating characteristics are very different from those of thin structures. The maximum radiating factor for thin structures is achieved at rather high frequencies (several KHz) while it is achieved at frequencies around only a few hundred Hz with sandwich structures. (see figure 16)

The figure shows the difference in acoustic radiation between the two types (thin and sandwich) that have been designed for a same helicopter. It must be noted that the sandwich structure radiates more than the thin metal structure at the most annoying frequencies (1 and 2 KHz). Conversely at low frequencies, the use of a sandwich structure is very favorable.

AN ALTERNATIVE APPROACH TO ENGINE RATING STRUCTURES USING MONITORING SYSTEMS

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SUMMARY - Optimising the engine size to the multi-engine helicopter's needs is a difficult process which has traditionally been carried out through the mechanism of the engine rating structure. However, the rating structure has many limitations and does not allow the best potential use of the engine to be realised in service. The introduction of the micro-processor based Engine Monitoring System permits a re-evaluation of the rating structure and also the presentation of limitations to the pilots. By using EMS it should be possible to achieve a better relationship between the demonstrated capability of the engine as shown in the Qualification Programme and the authorised release for in-service use. This may be regarded as a first step to be followed later by a change of the qualification test to a more representative form with the EMS giving a more tangible link between bench test and customer operation. This should allow better use of the engine to be made for the short time/high power requirement thereby allowing a more efficient engine performance at cruise conditions.

One of the most vexing problems continually faced by the engineers in the helicopter industry is how to optimise the match of the engine (and hence the power capability) to the multi-engine helicopters needs.

The technically elegant way of achieving this is to fit a larger number of smaller engines but there are good reasons why this way is not being followed such as increases in cost, weight and maintenance demands. It is likely that the optimum number of engines for most helicopters will remain at two with three being used only occasionally.

Whilst it is as well in discussing a subject such as this to avoid any direct association between what it is right to do and the methods by which it is achieved, nevertheless it may be that the time is right to consider an alternative approach to the traditional engine rating structure for helicopter power plants now that we have the Engine Monitoring System as a viable weapon in our armoury.

It is proposed in this paper to examine this subject in 5 sections:-

1. The rating structure and its purpose
2. The disadvantages of the traditional rating structure
3. Consideration of the difficulties in changing the rating structure approach
4. The integrity of engine monitoring systems
5. An alternative approach

1. THE RATING STRUCTURE AND ITS PURPOSE

The objective of a rating structure is to define a set of power/time limits which enable the helicopter to achieve its optimum role performance and, at the same time, ensure an acceptable level of airworthiness and safety.

It has been long established practice to test engines to specific ratings which are mandatory limits on the pilot for time of operation at specific power levels. These form a practical but approximate method of limiting the rate of cumulative damage to the engine in order to achieve acceptable lives. The format of the testing required goes back to the piston engine era and has been modified but not fundamentally revised since the early days of gas turbines.

The rating structure in use by the British Ministry of Defence is shown in fig. 1 and is a typical example with minor variations of that employed by all the leading Aviation Authorities, both military and civil, throughout the world.

For any engine, its ratings are validated by a series of tests carried out to strictly defined rules laid down by the authorities. This is called the Certification, Qualification or Type Approval Process.

Unfortunately the conditions specified do not always equate with the needs of a helicopter and authorised variations have to be built into the processes with the result that it becomes irrevocably linked with the particular installation.

A. TWIN ENGINE OPERATION

Maximum Continuous	-	The highest rating of the engine which may be used continuously in flight
Maximum 5 minutes	-	The maximum rating of the engine which may be used for a duration limited to 5 minutes per flight

B. SINGLE ENGINE OPERATION

Maximum Contingency	-	The maximum rating of the engine which may be used for a duration limited to $2\frac{1}{2}$ minutes during take off and landing
Intermediate Contingency	-	The maximum rating of the engine which may be used for a duration of 1 hour during an en-route failure of the other engine.

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Fig. 1 - Rating structure terminology. Twin engine rotor craft.

2. THE DISADVANTAGES OF THE TRADITIONAL RATING STRUCTURE

The gas turbine power/life characteristics are basically incompatible with the balance of the various modes of power requirements of the twin engine helicopter.

Some of the particular aspects are described as follows:-

Power requirements - The power demand of a rotary wing aircraft varies typically with air-speed as shown in fig. 2.

As well as requiring to be low cost, light weight, economical and reliable, the engine has to provide over a range of ambient conditions, levels of:-

		Power factor
i)	Efficient cruise power	100
ii)	High power for take off and landing	170
iii)	Higher power for en-route flight after loss of one engine	200
iv)	Higher power for transient to forward speed and climb after loss of one engine in the hover	300
v)	Higher power to maintain height in the hover after loss of one engine	340

The dilemma that faces the designer trying to cover this widespread of power is all too obvious and if he decides to provide enough power within the traditional rating structure to meet (v) then the engine suffers from being too heavy and having a too high fuel consumption to be optimum at (i).

Operational environment - The certification programme has to cover the wide operational envelope in terms of ambient pressure, temperature, humidity, air cleanliness etc.

The resulting effects on engine speeds, temperatures, pressures, vibrations and life are expected to be evaluated in a programme which is already disproportionately expensive compared to that of the helicopter.

The programme has to cover the full corners of the envelope demonstration, i.e. on the hottest day, at the highest altitude with the most inept pilot handling a low performance engine, made in the worst material, to the biggest tolerances with the most inaccurate instruments; and of course, it is assumed that all these factors can be present at the same time - a probability that would be regarded as extremely remote in most walks of life.

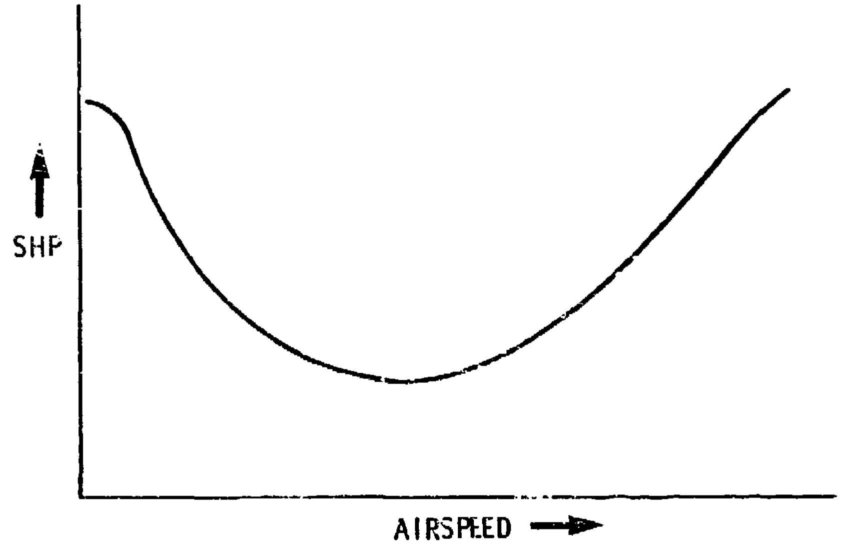

Fig. 2 - Typical helicopter power requirement

The power requirements for a specific operation with respect to day temperature are shown on fig. 3. It can be seen that the power requirement on a helicopter rises slowly with AIT whereas the power available falls sharply over the same range.

This is another basic incompatibility which has to be catered for in the approval programme.

The dilemma that results is, if the engine is tested at high ambient temperatures, it does not produce the power to test the transmission. If the engine is tested to the cold day power level it is not exposed to its maximum turbine temperatures. As usual it is a compromise between the two with auxiliary tests being run to make up the short falls where necessary.

Additionally, there are the installational aspects which override some of the engine limitations and by impinging on the supplementary approval tests as they do, start to make the approval programme specific to a particular type of aircraft.

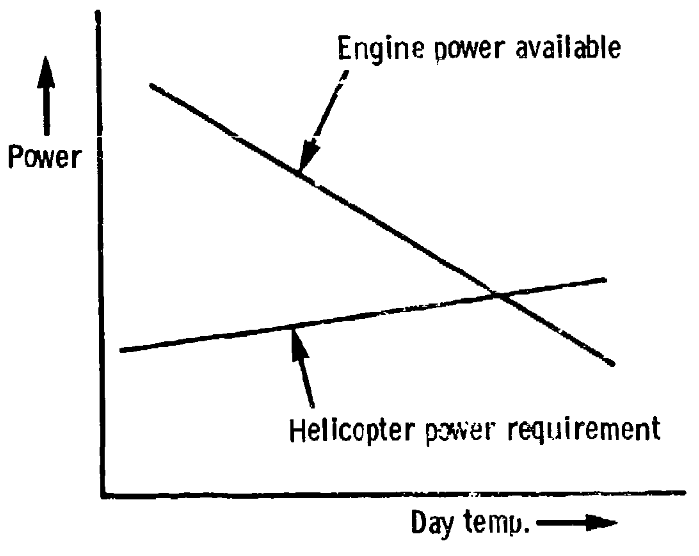

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Fig. 3 - Typical helicopter vs engine power

Emergencies - The concept of a rating is a "power" level usage associated with a time which enables helicopter performance to be scheduled in a flight manual and checked by flight test.

This assumes that all operation is normal - i.e. that the helicopter does not enter the "avoid areas"

where it will be unable to fly on one engine. Hence in the UK the use of the word "contingency" rather than "emergency". What the contingency rating does is to reduce the avoid areas, but it does not unfortunately, eliminate them. It now seems that helicopter designers are becoming interested in eliminating the avoid areas altogether by calling for higher power levels to deal with the true emergency.

If we are going to provide for the ability to cover emergencies then it has to be ascertained that the engine is capable of doing so. It is worthwhile quoting here from the Civil Aviation Authorities "Bible" in these matters - the BCAR.

"Definitions of power/thrust in terms of usage and duration (and the use of these to form the basis of certain flight manual limitations) are not intended to remove the pilots right to judge whether and to what extent such limitations may be ignored in emergency conditions."

It seems to us, as engine designers, that with the better tools we now have at our disposal we ought to be able to ease the interpretation of "emergency conditions" by the pilot and allow him to devote his energies entirely to making safe manoeuvres without the worry of what is happening to the engines in the process.

Applications - One of the major problems faced by the Technical Departments of Aero Engine Companies is how to cater for the wide variety of rating structures that current operators and potential customers are seeking.

This range has been extended recently by the proposal to insist on full category A capability for all medium twin engined helicopters.

The range covers such diverse items as:-

i) Anti submarine warfare operation, requring long period hover ratings for sonar dunking and an emergency power level for fly away following single engine failure.

ii) Armed attack warfare with high speed dash capability and nap of the earth power fluctuations.

iii) Recovery and ferry of aircraft with single engine failures from difficult take-off sites such as oil rig platforms or the surface of the sea.

iv) High altitude surveillance in mountain ranges with ability to elude sniper fire-arms by agile manoeuvrability.

These are just some examples of the latest proposed uses of the ever expanding role of the helicopter and they all, in their turn, produce small variations in the qualification requirements for the engine. These can generally result in a repeat type approval test unless one is extremely lucky and has covered the requirements inadvertently during earlier programmes.

Pilot observance of limitations - With the traditional rating structure the "red lines" on the cockpit instrumentation for engine temperature and speed can usually number four.

Four red lines on a cockpit instrument may sound acceptable but it should be remembered that engine instruments are not primary flying devices and are therefore restricted to a small size with a small scale for a large range. The actual observance of these limitations by the pilot is very difficult and the suspicion exists that the discipline employed in the observance of engine limitations may leave something to be desired.

Since the clearance of the engine limitations forms a significant base of the engine approval programme, the disparity between the effort that is put into selecting and clearing these limitations and the way this information is displayed and may be observed by the pilot is extremely frustrating to engine designers.

General - Whilst the five specific subjects dealt with above constitute anomalies and irritants which emanate from the rating structure, the general problem which swamps all of them is the disproportionate hot section factor damage which the current rating approval process accumulates in relation to other factors such as low cycle fatigue counting. The seriousness of this problem manifests itself not in some unreliability in service effect (such as would be the case if the LCF programme had been of poor quality) but in the far more fundamental form of the engine being basically the wrong size for its applications and it is this subject that is dealt with in more detail in the next section.

3. CONSIDERATION OF THE DIFFICULTIES IN CHANGING THE RATING STRUCTURE APPROACH

The clearance of a rating structure is achieved by what is called the type test in Britain and the model qualification test in the US. It is 150 hours long and has been in existence since at least the 1950's. It is beloved of Airworthiness Authorities and bears absolutely no resemblance to actual customer use.

On the other hand, it is regarded as a very sound and dependable yard-stick as to the measure of an engine's suitability to enter service and any suggestion to alter it is usually met by trenchant resistance.

As stated at the end of the previous section, its unrepresentativeness is mainly concerned with its high use of hot section damage, be it of a conventional creep usage form with solid turbine blades or of a more thermal fatigue bias for cooled blades and static nozzle vanes. For the purpose of this argument the hot section factor damage will generally be referred to as creep.

What could be changed initially is not the type test itself but the relationship between the demonstrated creep capability during type test and the authorised release life/power figures which come from it. This certainly ought to be a short term target with perhaps a slow change to a more representative type test to follow. Any change to the type test structure would probably require a doubling up of new and old styles for a period to demonstrate consistency of standard and while this is happening, programme costs would probably increase.

The long term returns however, should be very beneficial in relation to reduced cost of ownership in service and a more efficient use of some of the world's dwindling oil supplies.

The most obvious change that could be made to the type test is to impose the condition of a torque limitation for the normal temperate conditions under which the helicopter will operate. As was said earlier there is no recognition of this in the regulations and if it were employed it would allow a greater proportion of the creep usage in the test to be devoted to the tropical operation and thereby allow a higher engine temperature to be cleared for that part of the helicopters use. Fig. 4 illustrates this point simply.

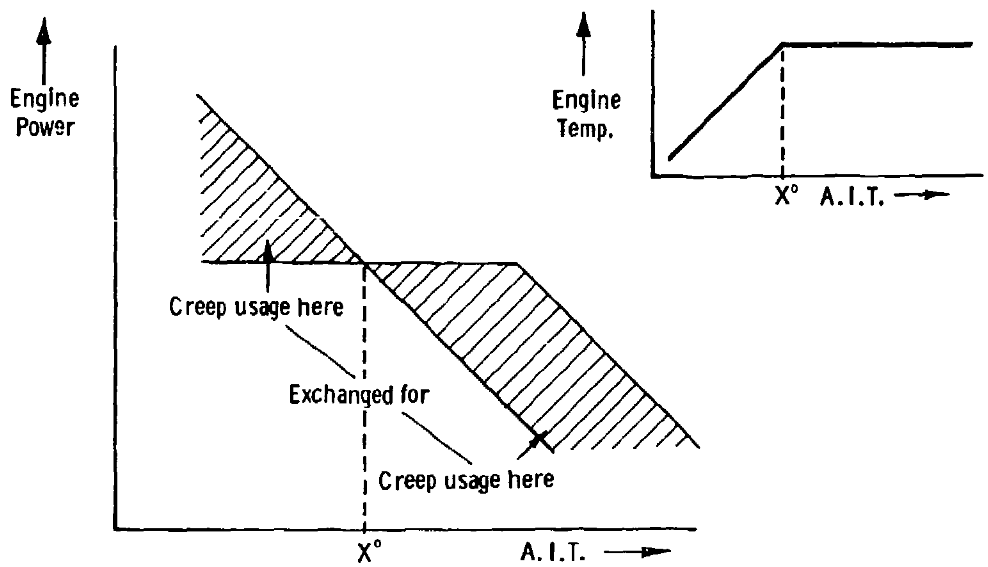

Fig. 4 - Type test with superimposed torque limits

The problem here is the accurate recording and re-allocation of the creep capability and hence we come to the Engine Monitoring System. If the proposed system with its programme algorithms for hot end usage (as well as other functions) is run on all the endurance testing during the development programme, then a sound basis of demonstrated creep capability is established which allows a degree of flexibility to be used in the type of release which the qualified engine can be given for potential customer use. In other words by compiling a data bank of the hot section damage capability for the engine, a new application with its power requirements can be evaluated against it to see that the new rate of damage and top power requirements are not incompatible with what has already been demonstrated.

It is interesting to consider what happens today when a rating is exceeded, say on an en-route failure either in duration or temperature. The pilot reports it (let's be kind!) and the operator's engineers consult the maintenance manual. Invariably there are no clear instructions on what to do, so they ask the manufacturers. The manufacturer's Service Department ask the Stress Engineers who generally say it's acceptable. (There are, after all, no definite rules to say how many times an OEI rating may be used in an aircraft life). The engine stays in service and the aircraft flies on.

If it happens another eight times, say, the answer may be different but this stems from two

reasons - one, the incidents have been recorded and two, the Stress Engineers have knowledge on the strength of materials.

Recording and storing knowledge are things that computers can do so why can't this process be dehumanised and put on to a computer?

The above function should be taken in context with the other necessary tasks undertaken in an EMS. These are limit exceedence, rotative LCF counting and thermal fatigue monitoring. All are a measure of the usage of the engine and become more cost effective if integrated in one unit. Additionally the pure health monitoring aspects of performance trending, vibration data analysis and lubrication system status can be added to provide a comprehensive monitoring system.

LCF counting is not covered extensively in this paper as it is already a practiced art-form in the industry. The full EMS does allow however, a higher degree of sophistication to be adopted on this subject and an illustration of the possible benefits is shown on fig. 5. The three lines show the progression in authorised, hourly life possible from a demonstrated cycle fatigue programme:-

i) The lowest line represents the traditional approach with little or no knowledge of the type of operation the customer is using. e.g. Mission spectrum and/or typical ambient conditions.

ii) The middle line represents the modern military approach where a mission profile with severity factors applied is defined at the onset of a qualification programme to form the basis of the component lifing philosophy.

iii) The upper line represents the release life possible from actual recording of service type operation at temperate sea level conditions.

It can be seen there is a three-fold advantage gained in the actual release life possible (the figures incidentally are real from a current Rolls-Royce programme).

From the fatigue point of view then, the rating structure and the type test are almost irrelevant and the cyclic qualification programme stands on its own. It is fundamentally hot section usage, limit exceedance and pilot observance that this argument is about.

4. THE INTEGRITY OF ENGINE MONITORING SYSTEMS

For engine monitoring systems to become group 1 equipment,(in other words, the release life and hence the airworthiness of the engine is dependent upon the results which the EMS provides,) the integrity of these systems will have become very high and this will take sometime to achieve. Initially it seems reasonable that they should be fitted to aircraft in group 2 equipment role, the engine released to traditional methods and the EMS monitored to ensure that it is producing valid data bearing at least some resemblance to the estimated life usage produced by manual methods.

We should certainly quote our own experience to date. Rolls-Royce has a helicopter operation in its flight test facility and an EMS has been under evaluation on this helicopter for a period of 5 years. This system was MOD furnished equipment to Rolls-Royce and was one of a number of projects supported by MOD during this period. The others were all direct service trials and the benefits accrued are available for examination. The system is a fairly simple first generation device comprising multiplexing Data Acquisition Units and Cassette Recorders. All computation is carried out off-line at ground replay stations. With this equipment we have shown that good valid data can be acquired in the harsh environment of a helicopter. We have also shown that such equipment needs intensive development if it is to become totally reliable.

Computer programmes for LCF, creep, performance trending and vibration monitoring have been developed and proven and some details of the configuration and the results are shown in the following figures:-

Fig. 6 shows the equipment configuration in the helicopter.

Fig. 7 shows the flow chart for LCF usage calculation.

Fig. 8 shows the flow chart for creep usage calculation.

Fig. 9 shows the recording of temperature versus time over a 400 hour flight programme.

Fig. 10 shows the recording of torque versus time over this 400 hour flight programme.

Fig. 11 shows the idstribution of creep usage over different types of flying carried out during the 400 hour flight programme.

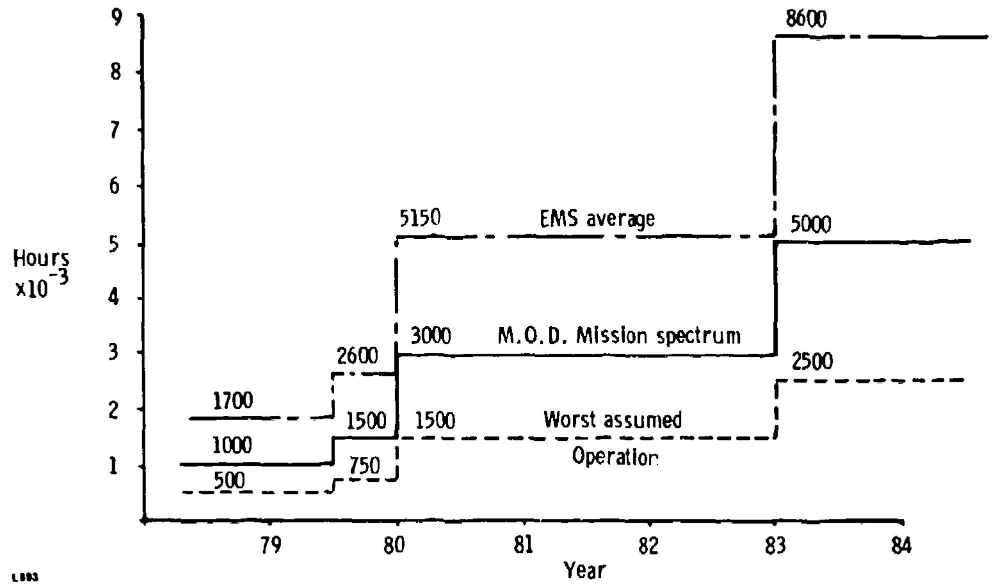

Fig. 5 - Benefits to LCF using EMS

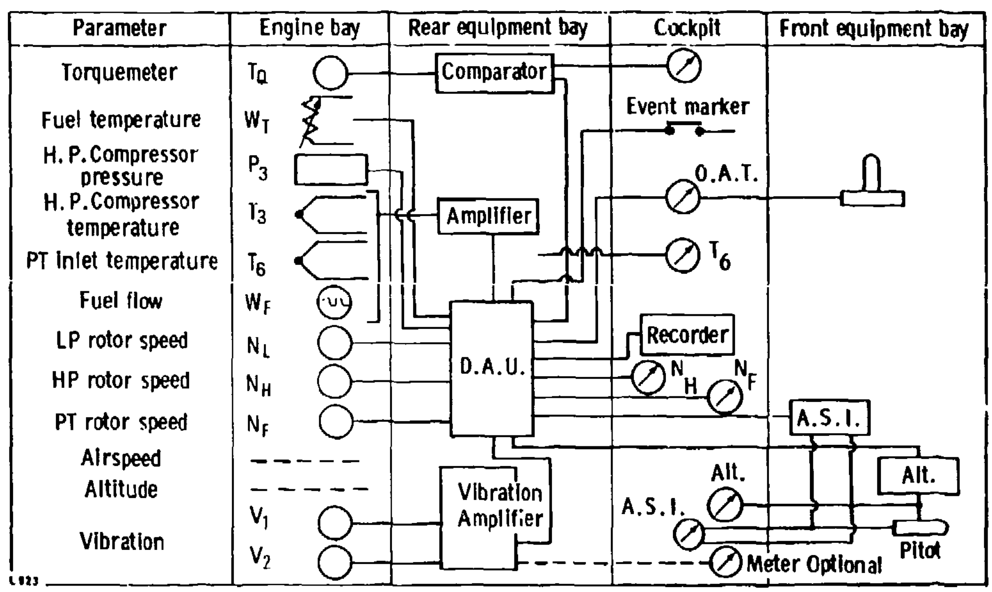

Fig. 6 - Early engine monitoring system

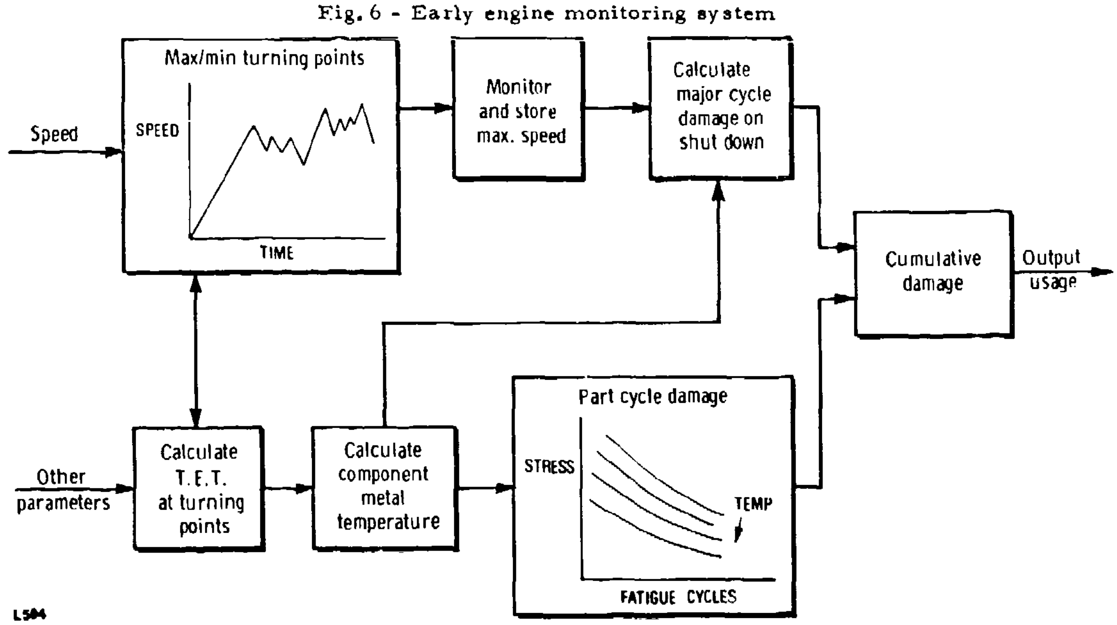

Fig. 7 - Low cycle fatigue. Calculation of cumulative damage.

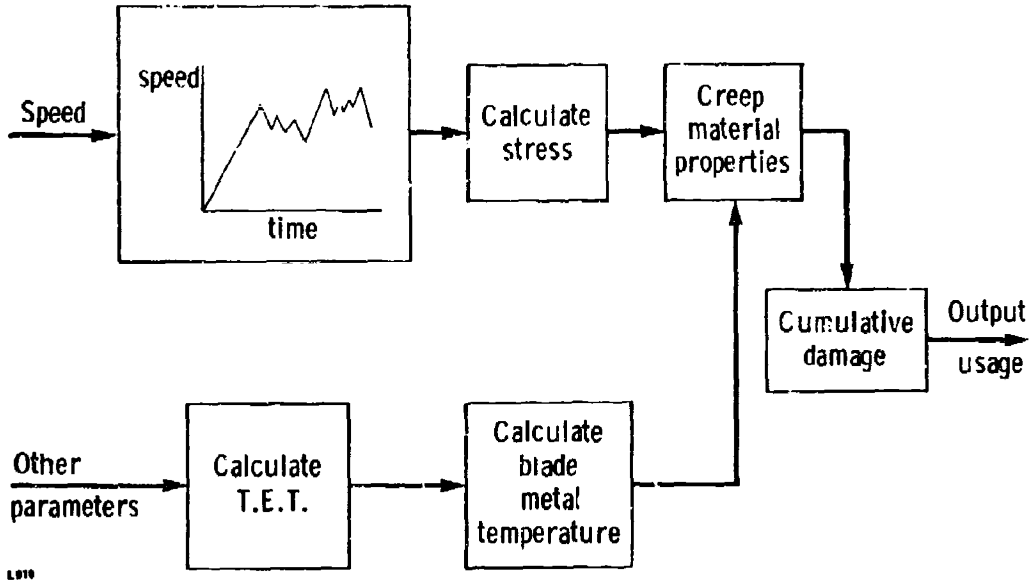

Fig. 8 - Turbine blade creep. Calculation of cumulative damage.

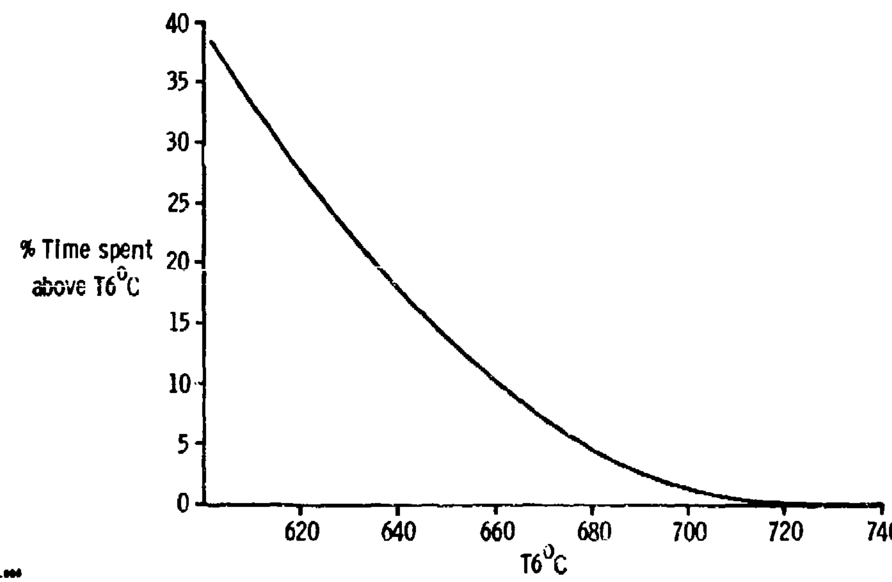

Fig. 9 - Typical time temperature analysis

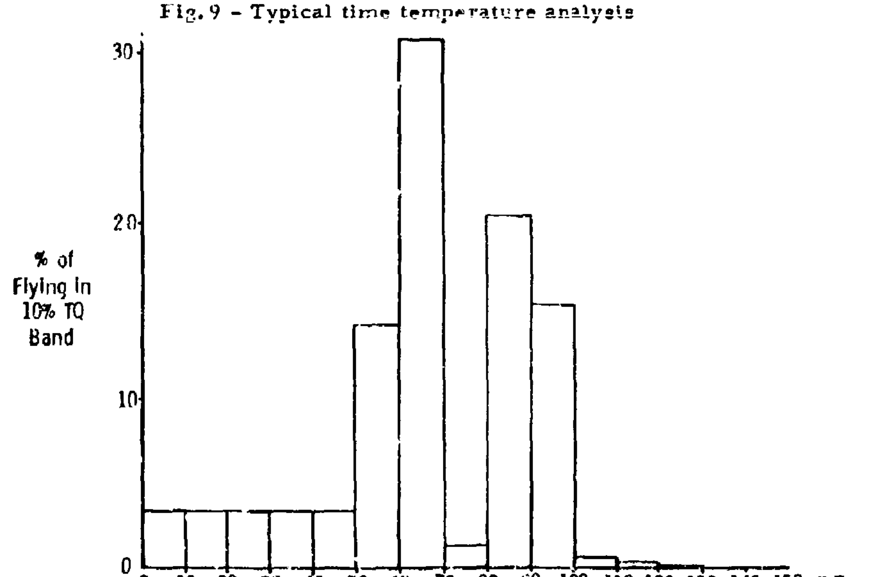

Fig. 10 - Typical time torque analysis

Flight Description	No. of Flights	Flight Hours	% Creep Used
Mission	40	72.4	0.239
Development	29	40.6	0.246
Crew training	2	3.1	0.004
Ground run	18	8.7	0.003
Air test	2	1.2	0.020
Tracking	5	4.1	-
Miscellaneous	48	80.2	0.725
Total	144	210.2	1.237

Fig. 11 - Typical blade creep usage (200 hrs)

Some of the surprising aspects were:-

i) The smoothness of the time temperature curve.

ii) The discontinuity in the time torque curve.

iii) The wide variation in creep usage for different types of flying, e.g. if only the prescribed mission was flown then 50% creep usage would allow 15000 hours flying of this type. However, the rate is three times this for "miscellaneous", but as the programme was flown by test pilots, we believe we can still afford to be relaxed on the subject.

Based on the experience described above, together with Rolls-Royce's experience on electronic control systems over 20 years and the fact that the industry is apparently poised to change over to digital electronic control systems en-masse, there is now a general feeling that the integrity question concerning monitoring systems can be faced with confidence and that only extensive field experience can progress the state-of-the-art from this point onwards.

5. AN ALTERNATIVE APPROACH

The basis for an alternative approach is a long held belief that much greater utilisation of engine components could be achieved if a more accurate count of hot end damage occurring in actual engine usage is made.

Another factor forming the basis of this proposal is that when the pilot is in a critical situation, the last thing he wants to be bothered with is engine limitations. He is there first and foremost to fly the machine and his eyes should be "out" as much as possible not "in".

The approach is therefore two-fold; maximising economics and minimising workload.

The proposal is set as a bold step representing an ultimate position with the express purpose of challenge, stimulation and discussion rather than dwelling on a tortuous path of how actually to get there.

Qualification - The programme should be aimed at establishing a temperature/time cumulative damage bank for the engine components compiled from actual engine running. There must be dedicated creep and/or thermal fatigue evaluation testing using components of known strength from known batch manufacture. This testing will be exhaustive and must result in failures to be effective. (Some engines actually experience component creep failure during normal running but Rolls-Royce engines are not normally in this favourable position!) The new temperature measurement techniques such as radiation pyrometry will be employed. Two milestones will be achieved by this testing. Firstly, the critical part of the creep curve will be established for the real environment of the engine and secondly it will be demonstrated consistently that the failures are "soft". The by-product will be that the early warning failure detection systems will be tested and developed for real. The creep usage meter will be monitoring all the testing and a good relationship will be established.

This testing will be supported by the currently employed thermal shock and simulated mission tests to establish the degree of inter-action of creep and fatigue. We must demonstrate clearly to the authorities that we have a sound knowledge of the creep capability of the engine - this is the only way to be able to reduce the severity of the safety factors currently employed and start to achieve the

objectives.

The qualification test itself will be changed to a mission based schedule and will take into account a likely aircraft torque limit over the lower ambient temperatures. The test will be of longer duration than the current one, probably 3 or 400 hours and will be aimed at using at least $\frac{3}{4}$ of the potential life of the hot components. The engine temperatures will be chosen to achieve this figure over a spectrum or air inlet temperatures. This spectrum should cover ISA to ISA+25^{o}C SL in four equal steps, the engine temperature being at maximum for the top two AIT's and reduced according to the engine torque limit for the others.

The composition of the mission will be based primarily on twin engine power levels but at intervals to be agreed, simulated single engine power levels will be incorporated varying according to engine failure, occurring at different points in the mission. The frequency of the single engine cycles should be commensurate with typical figures for engines in their infancy in service.

The test would continue with normal problems and failures being repaired as in service but with major failures of Design or Quality rendering the test void.

Allowance for performance deterioration would be built in by raising the temperatures in the torque limited stages accordingly.

At the end of the qualification test and the supplementary endurance testing, the creep accumulation will be evaluated and a case presented to the authorities for an agreed initial creep release to service. If things have gone well, then approximately 2/3 hot section life could be claimed. It might be argued, however, that if too high a factor is achieved then the estimates were too low in the first place, but that is just one of the frustrating vagaries of this business!

For further supporting evidence it is believed that the qualification test engine should be run on to failure, in an accelerated form if necessary to minimise the cost.

The last point about qualification is the establishment of the absolute maximum temperatures and speeds at which the engine can safely operate. These should then be protected by the automatic limiters of the engine control system and the resulting power should be available to the pilot in an emergency. As long as the damage is being recorded, the appropriate maintenance action can be taken after emergency power levels have been used.

Certification - The engine will be certificated with a carpet of characteristics (obviously on a card deck) showing the relationship of power versus percentage creep life usage per hour for all ambient conditions. It is believed that this is really all the aircraft designer needs to know to select an engine in order to compose his flight manual around it.

Operation - The pilot will be relieved of any need to observe the engine temperature and speed instruments. He will need to be warned of the fact that he is employing a high rate of hot end usage and this warning needs to be progressive. The computer can be programmed to do this but how it is presented is not for us to say. Pilots tend to have strong views on this matter!

The onus of monitoring the usage should be on maintenance personnel. The amount could be displayed in the cockpit to be read at appropriate intervals or more likely in Civil operation, to be fed out on to a ground printer.

If the computer fails in flight, advisory procedures for safe operation need to be spelled out in the flight manual but it should not be regarded in any sense as a serious problem and there should be no need to abort the mission unless it is of a highly critical nature.

A full "on condition" maintenance approach is compatible with this system and that concept is now gaining wide acceptance in the industry.

The engine is withdrawn for appropriate action when the authorised hot section factor is achieved.

Hardware - For military machines some aircraft contractors envisage a full integration of the EMS with the main aircraft computer system as shown in fig. 12. A visual display unit with interrogation capability is available to the pilot in flight and to Maintenance during servicing periods.

For Civil machines the system should be a stand alone unit primarily to provide option for fitment to the customer.

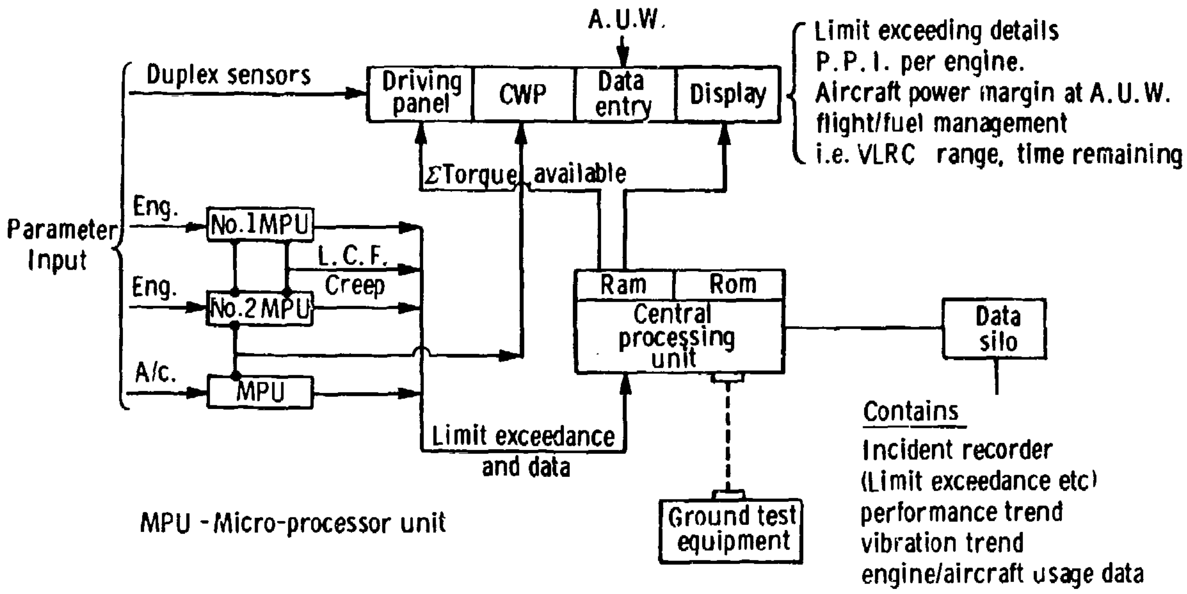

Fig. 12 - Integrated helicopter monitoring system

CONCLUSION

This paper is aimed at basically trying to achieve a better use and hence return for money of todays costly engines.

It is believed that it is the certification test that sets the size of an engine for a helicopter application under the current rules and that the disparity between this and the need to have efficient fuel usage in service needs to be constantly questioned, always within the broad envelope of safety that is termed "airworthiness".

There should not be a tacit acceptance that the present rating structure and pholosophy is sacrosanct for ever-more. It is one of the engineers basic functions to always question the status quo to see if it can be improved and this paper is offered against a pursuance of that function.

ACKNOWLEDGEMENT

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DISCUSSION

Unknown Questioner

Do you believe emergency ratings are appropriate for civil applications and can such ratings be believed from an approval standpoint without demonstration?

Author's Reply

I am basically against the rating principle but am proposing that, as an alternate, a monitoring system be used to gauge remaining life of the engine.

Unknown Questioner

Could you comment on the relative importance of performance monitoring as compared to creep monitoring, particularly for off-shore applications?

Author's Reply

Both are required in order to determine the remaining life of the engine. The real difficulty is to convince the certifiers that such trend monitoring is a valid indicator that the engine will produce the demanded emergency power. If such power must be demonstrated, the whole concept is self-defeating.

COMPONENT RESEARCH FOR FUTURE PROPULSION SYSTEMS

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ABSTRACT

A review of the factors affecting the helicopter market for the past, present, and future is presented. The trade-offs involving acquisition cost, mission reliability, and life-cycle cost are reviewed, including civil and military aspects.

The potential for advanced vehicle configurations with substantial improvements in energy efficiency, operating economics, and characteristics to satisfy the demands of the future market are identified. Advanced propulsion systems required to support these vehicle configurations are, in turn, discussed, as well as the component technology for the engine systems. Considerations for selection of components in areas of economics and efficiency are presented.

Introduction

There are many factors related to propulsion systems that strongly influence performance of helicopters. Reviewing each of them and their impact on the past, present, and future helicopter market is not practical in the context of this paper. Most significant, however, are the trade-offs among acquisition cost, mission reliability, and life-cycle costs. Beyond these factors, detailed assessments of potential advances become extremely complicated by end usage (military versus civil) requirements and by the escalating cost of fuel. Of course, each of these concerns finds root in the component technology needed for improved operating economics, and some of the critical issues are discussed here.

Outlook/Background

The growth in all sectors of aviation has been dramatic over the last 50 years. However, it is clear that rotary-wing aircraft have lagged this growth by as much as two decades. One reason is the increased difficulty in achieving controlled flight compared with fixed-wing aircraft, and another is the dependence on the development of different technologies, specially needed for helicopter components. Despite the lag, benefits derived from these special purpose machines have been growing at a significant rate since 1960. Figure 1 indicates the rate at which the activities have enlarged in North America. Also shown is a somewhat conservative growth projection for the next 10 years, and there is an equally active future projected for the helicopter industry world-wide. In the United States alone, the production rate for civil uses has grown from 300 units in 1965 to 800 in 1975. The 1981 figure is expected to exceed 1000 units. In most cases, the technology has been paced by military interests, particularly the United States Army. Now, the civil needs are emerging, and this segment will add strength by sharing in common solutions to most of the operational problems. As shown in Figure 2, the primary applications will be in forestry, public service, agriculture, resources exploration, and construction, in addition to short-haul, general transportation. Thus, there is a base to enlarge the research activity for both military and civil needs, and the generic aspects cover a wide spectrum of components from which substantial gains can be made. NASA's rotorcraft program evolved from autogyro research during the 1930's (at that time it was known as the National Advisory Committee for Aeronautics, NACA). From this pioneering effort, a close association later was developed with military rotorcraft R&D organizations. One of these strong ties was with the US.Army, resulting in a number of formal and informal cooperative efforts. The latest such agreement became effective in 1970. With it, NASA and the Army share resources to pursue research in areas of common interest, including all helicopter disciplines related to aeromechanics, structures, and propulsion. As a result, Army research groups, colocated with NASA, conduct inhouse and contracted efforts on all components of engines and drive trains, including materials. It is this association that is bringing into focus the commonality aspects of the civil and military interests as applied to fundamental problems and basic technology. To complement this joint effort, NASA formed a government-wide task force in 1978 to assist in formulating a long-range advanced rotorcraft technology program.

Systems Requirements

Civil operators continually emphasize a true, one-engine-inoperative (OEI) capability. They are unanimous in their endorsement of twin-engine helicopters, but they are unhappy with single-engine performance in the OEI mode. Ideally, these operators want a nonemergency situation in the event of an engine failure. As shown in Figure 3, their goal is to achieve a zero-rejected takeoff distance to enable operations in tightly confined areas, consistent with the attributes of helicopter systems.

Regardless of the user, safety and reliability continue to be the central issues. Although safety of twin-engine helicopters is regarded as an inherent advantage, unscheduled engine removals continue to frustrate all users. Dissatisfaction with support requirements and the attendant high costs are common. The component research programs must include emphasis on significant increases in the time between overhauls (TBO).

Maintenance costs associated with propulsion and drive-train systems are shown in Figure 4. A major reduction in maintenance costs is essential to the enhancement of helicopter operations. Beyond this need, recent advances in diagnostics and associated avionics are already finding their way to the market. With continuing progress in microprocessor technology, early identification of engine and drive-system problems, before they become serious, may provide the techniques for maximizing use of on-condition maintenance procedures. Figure 5 illustrates the need for significant improvements.

Electronic digital controls can provide large improvements over conventional hydromechanical systems in terms of simplicity, cost, reliability, and ease of operation. An example is shown in Figure 6. Time response characteristics of the total propulsion system (i.e., engines, control, and power transfer mechanisms) must be evaluated in a totally integrated mode. These evaluations must be conducted in systems which simulate, as closely as possible, the environment that will be encountered in service.

Powerplants and Component Thrusts

Since the advent of turbine engines for helicopters, a dramatic increase in load-carrying capability has evolved. Almost all of these turboshaft engines were developed for military helicopters, and each is based on technology derived from military-sponsored development. Even so, it is expected that the next two decades will find a growth in the civil markets to the extent that the number of engines produced will be about twice the military needs. This growth, combined with the comparatively high usage rate (2000 flight hours per year) for civil units, will substantially enhance operational evaluations of engine-related technologies. The civil demands for reliability, maintenance, and overall cost will be stronger than at any time in the past. The experience with large field samples will be invaluable.

Performance improvements have become essential, particularly for those future helicopters that will be designed for increased range and speed. Toward that end, perhaps more complex engine arrangements, using highly loaded components, recuperators, and variable geometry will be required. If so, we must be in position to compensate for the likelihood of higher initial costs with superior, fuel-efficient engine systems. Obviously the engine cycle must be improved in the partial-power regimes, shown in Figure 7, and emphasis must be placed on methodology to provide aerodynamic components that will produce a specific fuel consumption characteristic more nearly flat than in today's conventional cycles.

In an overall sense, the objectives of the NASA-Army propulsion efforts are to (1) improve engine and power transfer component reliability and maintainability, (2) reduce engine fuel consumption over the full range of operation, (3) improve environmental acceptability, and (4) reduce the cost of acquisition and operation. The highest priority program element is component design methodology, as applied to each of these areas. More specifically, achievement of the objectives will concentrate on the six key technology task categories shown in Figure 8. Each of these program elements has been reviewed by a broad spectrum of civil and military users and, despite the diversity of missions, there remains a remarkable unanimity on the areas in need of most immediate attention. Concerns relating to powerplants appeared to lead the list of priorities.

Although the current program covers all components of interest, this discussion will concentrate on those considered to have significant impact on the above thrusts.

Compressor

In a gas turbine engine, the compressor design and technology are very important choices because of the effect on the overall engine performance and arrangement. Compressor pressure ratio and compressor efficiency have a direct bearing on fuel consumption. At the same time, the compressor can have an influence on the number of turbine stages as well as the number of spools.

Because of the important role of the compressor in the overall engine, a joint Army/NASA program was undertaken with General Electric, Detroit Diesel Allison, and AiResearch for design studies of small, axial, centrifugal compressors to provide a basis for focusing future research on small compressors. These studies started with a forecast of the projected 1990 state of technology in compressors and engine systems. The projections forecast improvements in technology derived from existing work, from advances in materials processing and manufacturing methods, as well as improvements from advances in design techniques and computer aids. Based on these 1990 projections, parametric studies were conducted on compressor configurations for 2-, 5-, and 10-lb/sec flow sizes. Four staging arrangements (single staged centrifugal, staged centrifugal, staged axials, and staged axial-centrifugal) were investigated. Compressor pressure ratios from 10:1 to 40:1 were studied. The optimum compressor arrangements for 2, 5, and 10 lb/sec were identified in terms of efficiency, reliability, durability, maintainability, and cost (Fig. 9). In the 2 and 5 lb/sec flow sizes both axial centrifugal and staged centrifugals configurations have the best potential for advanced rotorcraft propulsion systems. In the 10 lb/sec flow size, the axial centrifugal compressor configuration appeared to provide the greatest potential efficiency at the high cycle pressure ratio.

Improved theoretical aerodynamic analyses, verified by detailed quantitive data, will help improve the understanding of the complex flow field for the advanced axial and centrifugal stages. Techniques for the calculation of the internal flow field in centrifugal compressor using three-dimensional viscous computational methods are being developed (Fig. 10). To date, we do not have analyses which can represent the actual flow conditions with reasonable computing times.

Recent advances in the application of laser anemometers, or laser doppler velocimeters, permit measurement of flow velocities and mapping of the flow field. In Figure 11 the covers have been removed from the compressor to show the laser beams crossing. Windows in the housing permit the beams to be directed into the rotating passages or into the diffuser area. Problems with seeding are delaying testing at higher speeds in small centrifugals, but we are hopeful of having a nonintrusive means of obtaining flow information in small passages where probes previously disturbed the flow. The laser anemometer will be an extremely valuable tool for developing an understanding of the complex flow areas such as the discharge region at the tip of the impeller. This area is most critical because it is here that the diffuser converts the high velocity into pressure.

Other improvements will involve variable flow capacity, where variable geometry is used in the compressor and turbine areas. An essential feature is the capability of a variable diffuser for the centrifugal compressor. This will permit the flow to be reduced while maintaining or increasing the pressure ratio and operating at a constant corrected speed. Consideration must be given to the mechanical design features required to implement this system in light of the gains to be achieved.

Beyond this, there will be improvements in small compressor performance with clearance control, reduced endwall and profile losses, and low aspect ratio blading, which is less sensitive to wear.

Combustor

As compressor pressure ratios are increased, future combustion systems for small turboshaft engines will be required to operate at higher pressure levels and increased inlet and exit temperatures, which will increase the need for better cooling or liner materials. Programs are ongoing at the Lewis Research Center on both approaches. One example is a plasma-sprayed ceramic on a porous metal substrate as shown in Figure 12. This approach will permit a significantly higher liner hot-side temperature and permit cooling the substrate with much less cooling air.

Both combustor and turbine life are adversely affected by nonuniform combustor exit temperature. At the same time, the need for improved fuel economy makes desirable the reduction in pressure drop across the combustor. These two goals are in conflict because temperature pattern is easier to control with larger pressure drop. Improved modeling and analytical techniques promise to reduce the cost of developing combustors, and there has been significant progress made in this area for small, reverse-flow combustors. However, such studies have indicated the need for additional research in fuel-injection methods and in primary zone analysis and experiment. One such investigation is illustrated in Figure 13.

With these two areas of research, it is expected that designers of future engines will have better materials, better cooling schemes, and improved techniques for selection of parameters to make trade-off decisions affecting combustor life and fuel efficiency. In addition, they can expect the relative development cost of combustor components to be reduced because of improved analytical tools that have been verified by experiment.

Turbine

Current research for the small turbines used in helicopter engines is directed at improved efficiency and higher temperature capability. The primary thrusts are for fuel efficiency and longer life for maintainability and life-cycle cost purposes. Materials research is primarily directed at this latter requirement with emphasis on coatings--metallic coatings for oxidation and corrosion protection and ceramic coatings for thermal protection and reduced cooling requirements.

One means of improving turbine efficiency in the small engines is to utilize a radial flow instead of an axial flow turbine. It is recognized as heavier and more bulky than its axial counterpart, but, as shown in Figure 14, it has potential for better performance at high pressure ratios. The radial turbine also is best suited to a variable capacity engine, which may offer an advantage in fuel consumption over a fixed-geometry configuration. To maintain design point pressure and speed, the nozzle area needs to be varied in some manner such as illustrated in Figure 15. Current analytical and experimental research is directed at understanding and quantifying losses due to the variable geometry to permit a realistic assessment of the fuel saving potential of such a concept. Figure 16 shows experimentally derived efficiency of a variable-area configuration over a wide range of flow at a pressure ratio of about 2:1. While these data do not include stator leakage effects and losses that might occur with high exit swirl, consideration must be given to these factors as analysis and experiment are continued. However, the reasonably constant efficiency is encouraging and shows that this concept has potential for a practical variable capacity cycle. At the very least, research along these lines will permit designers of future helicopter engines to incorporate high-work radial turbines in their engines with higher efficiencies than now possible.

As the technology advances for employing nonintrusive flow measurements, such as laser anemometry, measurements will be made to further the understanding of the flow in radial turbines. Three-dimensional analytical techniques, when verified, will provide the basis for further advances in both the efficiency and cooling of radial turbines.

Mechanical Components

Our present concern for fuel conservation and the need for better performance retention necessitate improved seals. Many engines use labyrinth seals for inner air seals. Generally, these seals employ one or more stages of knife edges and, in the higher performance engines, have shown a high leakage rate. To overcome this problem, face-contact seals sometimes have been used; however, they are pressure and speed limited and have excessive wear. Recognizing this shortcoming, a high-speed spiral-groove seal has been developed through joint Army/NASA efforts. It offers a solution and can be regarded as a step in technology for replacing both the labyrinth and face contact seals. As shown in Figure 17, this type of seal has shallow recesses in the running surface which cause a buildup of high pressures and prevent actual contact of the carbon face, except during start up. This seal has potential for application in both current and future engines.

Of all the seals in an engine, the gas-path seals have the greatest impact on performance. Tests have shown that an increase of 1 percent in blade-tip-clearance-to blade-span-ratio reduces the turbine efficiency by as much as 3 percent, depending on the type of the turbine design. The effect is shown in Figure 18. Generally, the gas-path-sealing clearances change with each engine condition such as idle, takeoff, and cruise. A joint Army/NASA program to develop a solution has been in progress for several years, exploring various ceramic high-pressure-turbine seal systems. One concept, which is currently under investigation, shows potential. It employs a low-modulus cushion or strain-isolator pad between the ceramic layer and metal substrate, depicted in Figure 19. The strain isolator pad allows the ceramic layer to respond to its own temperature gradient, independent of the thermal strains and displacements of the metal substrate. In conjunction with this system, clearance-control concepts involving selection of materials as well as rotor and outer structure configuration to maintain a fixed clearance, are under investigation. Variables also include thermal expansion rates, and the potential for using

engine air to heat and cool the mating rotor and stator. Also, blade tip treatment concepts are under investigation to prevent damage to the blade tips at contact with the case. These seals will become more practical as materials, design concepts and manufacturing methods are developed further.

In the area of shaft dynamics, there have been several Army, NASA, and industry programs to develop improved balancing techniques for rotors. In addition, a new computer code, which is capable of predicting nonlinear rotor dynamics, has been developed. This code allows investigation of transient rotor motion during adverse operating conditions, such as a blade loss with a rub. Previously, shaft behavior could not be predicted with available rotor dynamics codes. Damper concepts also are being analyzed, primarily to explore rotor systems that will be more tolerant to a large imbalance, as in cases equivalent to the loss of a blade or foreign object damage.

Research on bearings will continue to focus on optimizing design through improvements in materials and lubrication for higher-load capacity and longer life. Computer programs incorporating the latest design techniques are in various stages of development and verification. Speeds over 50,000 rpm are common, and a moderate increase is forecast for the future. Along with this emphasis, there will be an increased attention placed on noncatastrophic failure of bearings operating at higher speeds.

Conclusion

In pursuing these areas of research, we have highlighted but a few of the many component details that need special treatment. All of the components technologies, whether in engines or drive trains, are supported by an aggressive program covering all facets ranging from aerothermodynamics to materials and structures. It is hoped that the growth forecasted for the helicopter will provide added incentive to concentrate the needed resources in the critical areas identified here. In cooperation with the industry, NASA and the Army will continue to explore the components and develop the technology to meet the growing needs, particularly as related to reliability and life-cycle cost.

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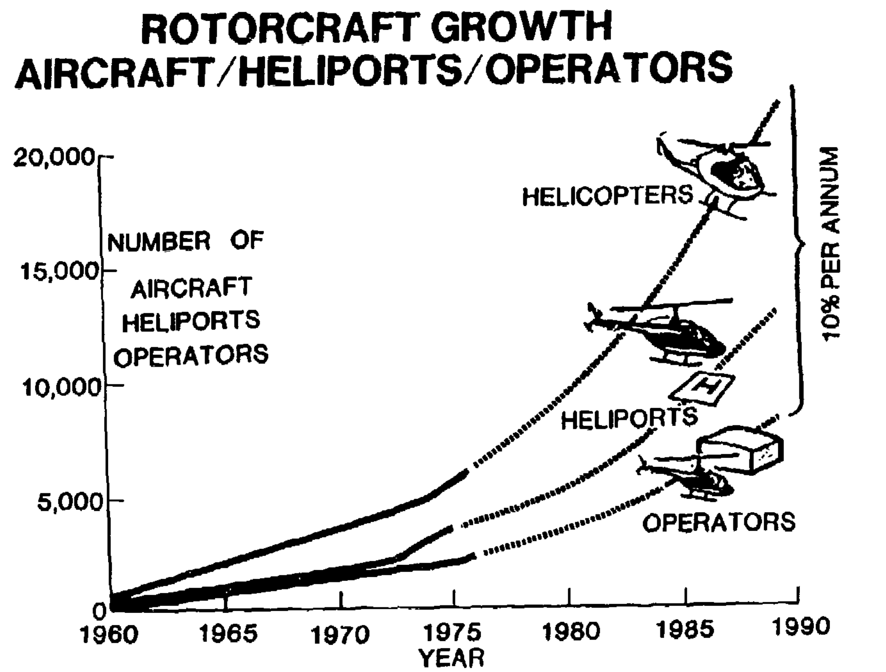
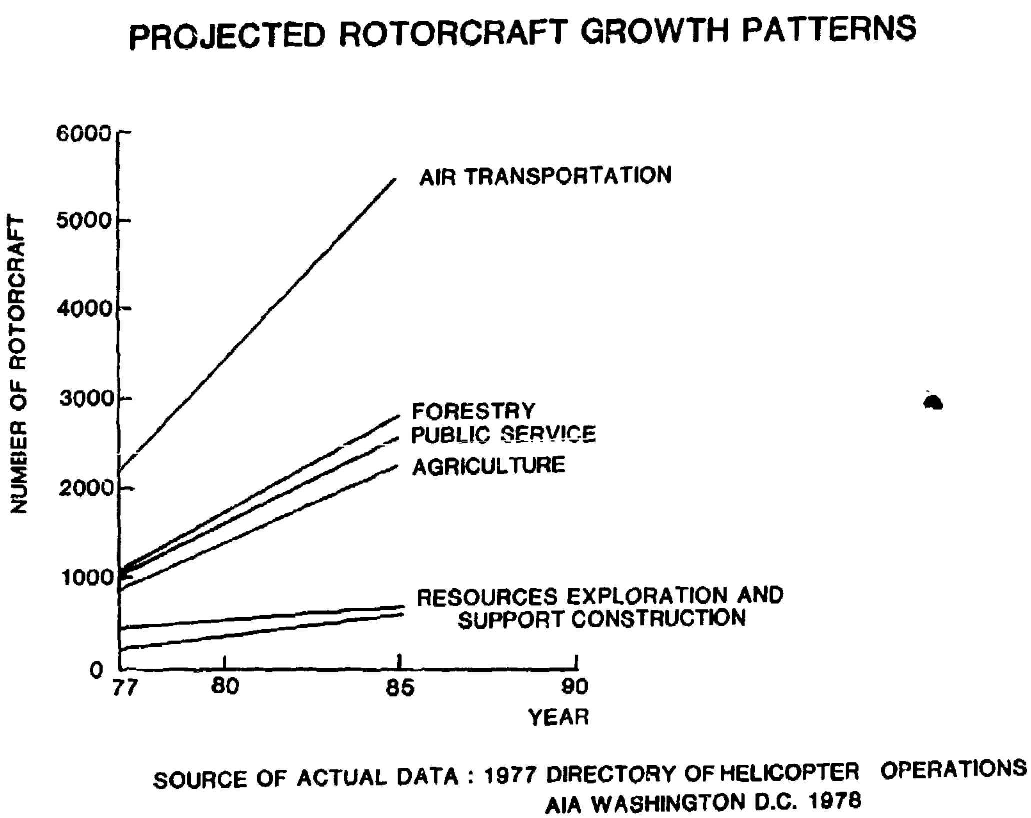


Figure 1



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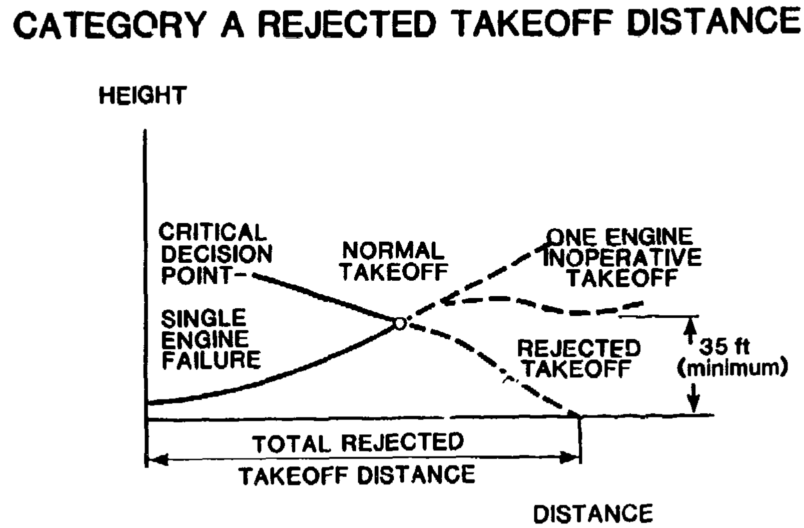

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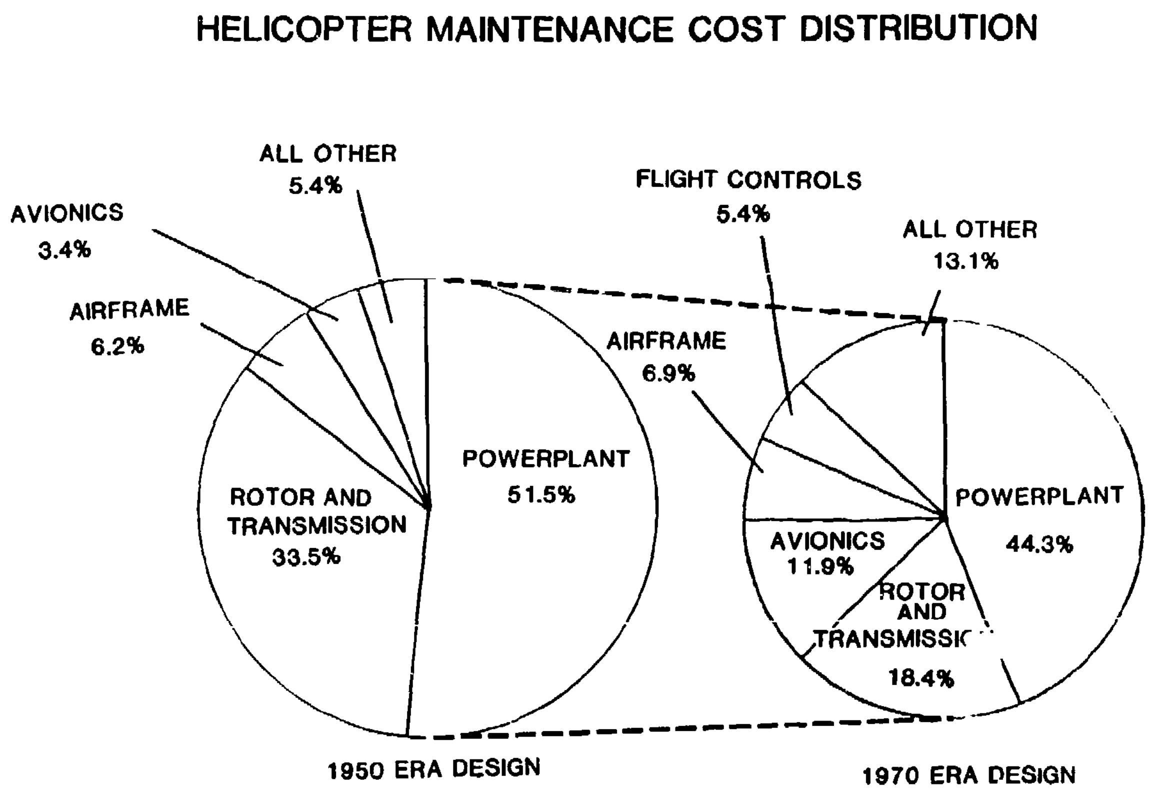
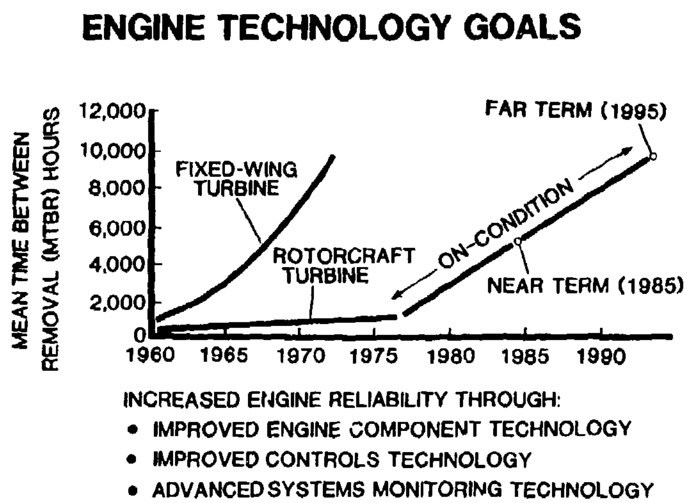


SOURCE:

1. 1950 DESIGN BASED ON ASSESSMENT OF UH-1H CONTAINED IN TR75-3
2. 1970 DESIGN BASED ON BOEING VERTOL YUH-61A DMC STUDY, DECEMBER 1975

Figure 4



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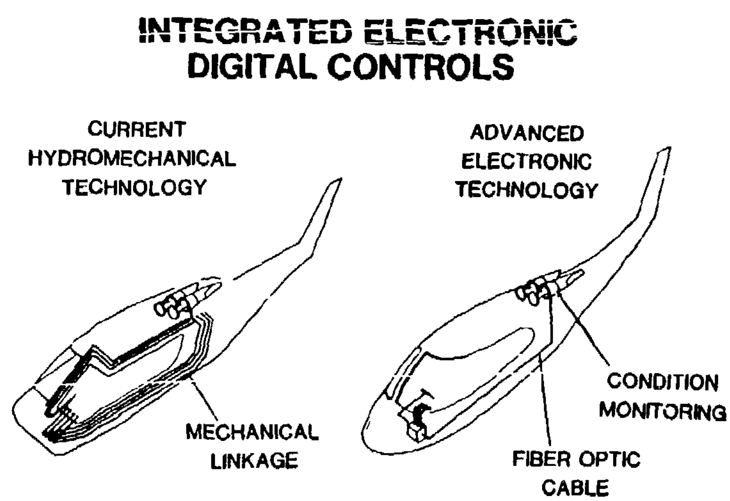

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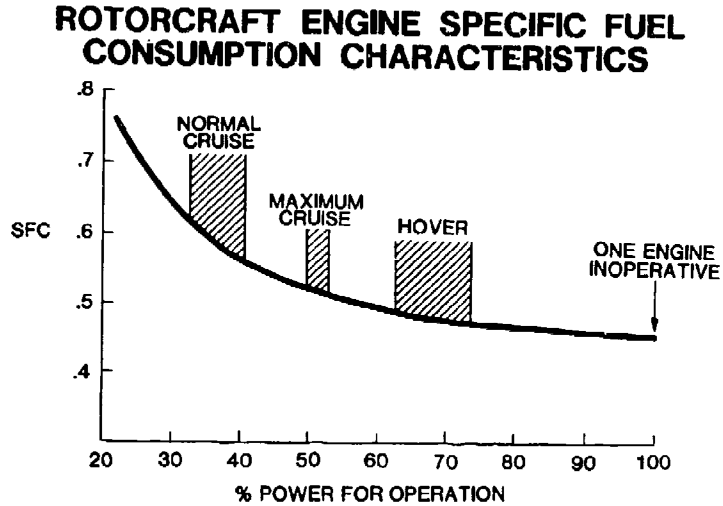

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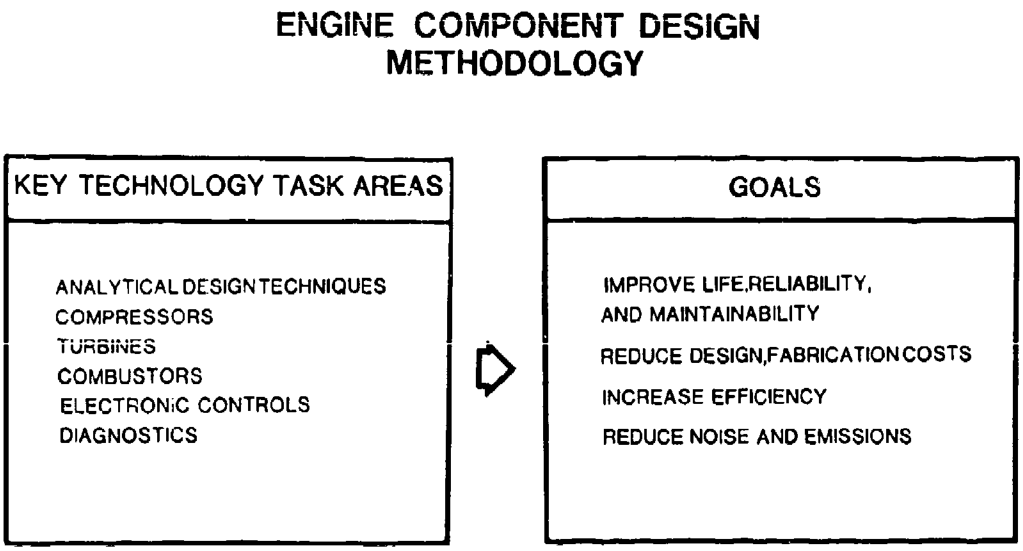

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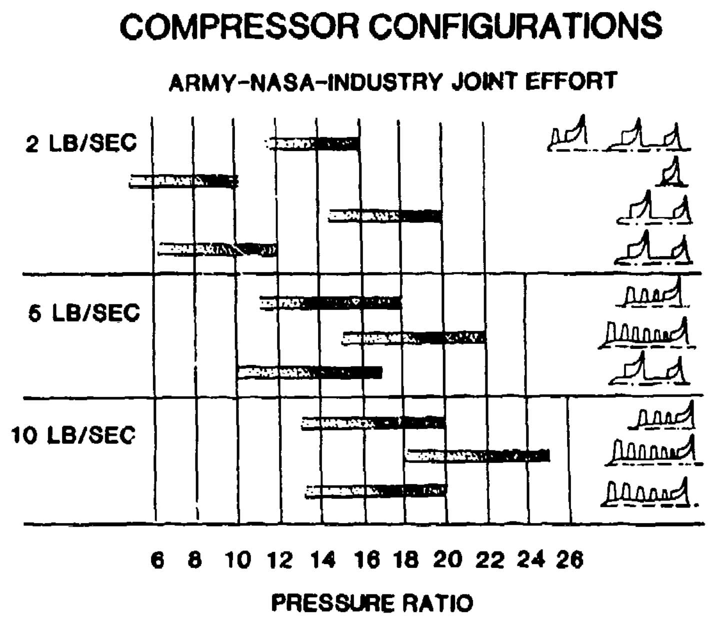

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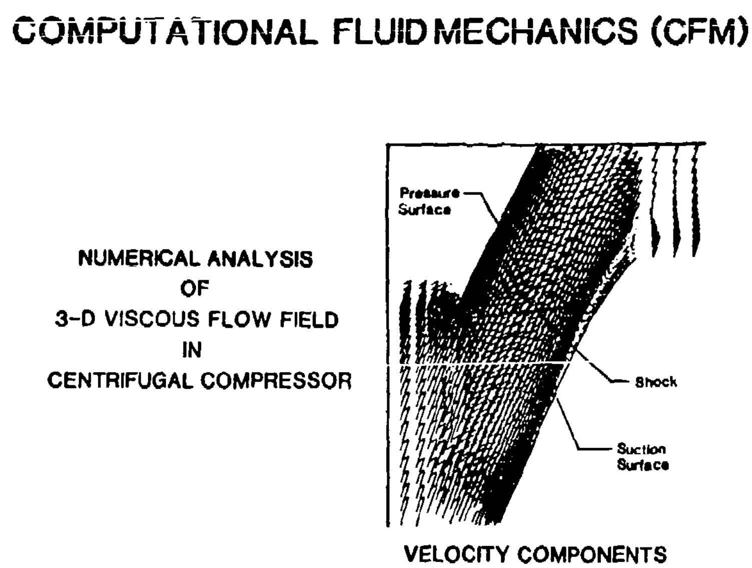

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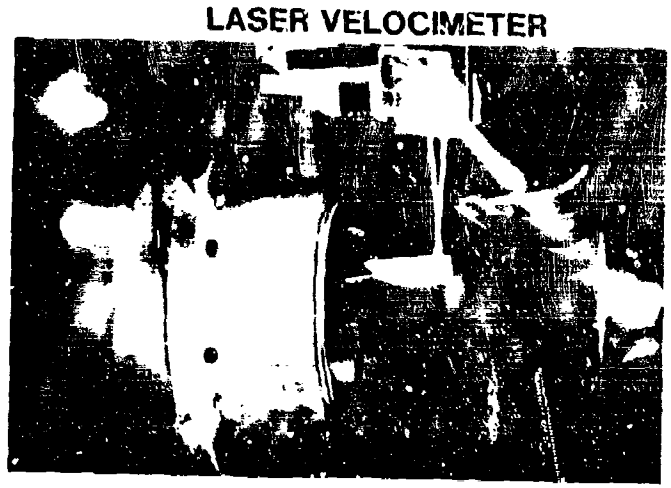

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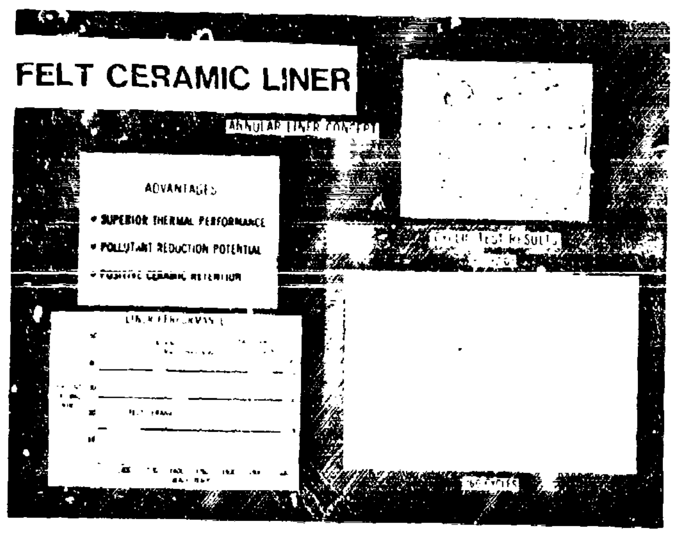

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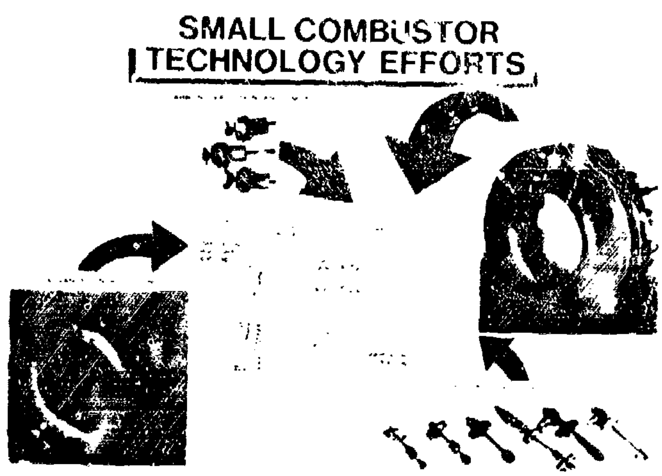

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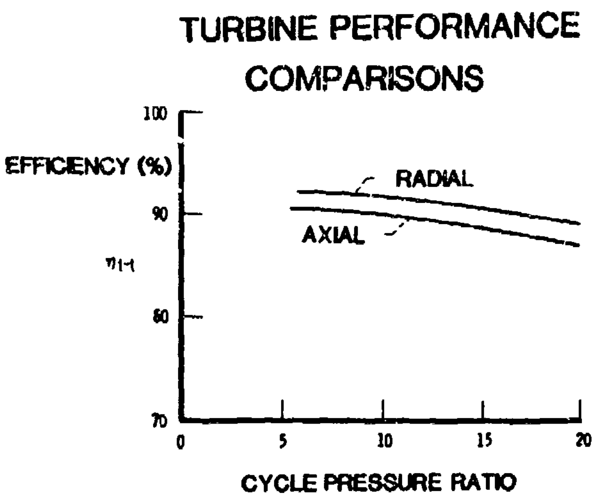

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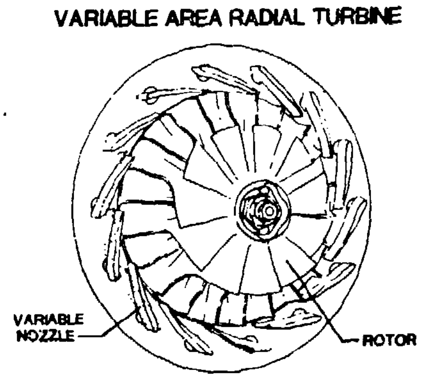

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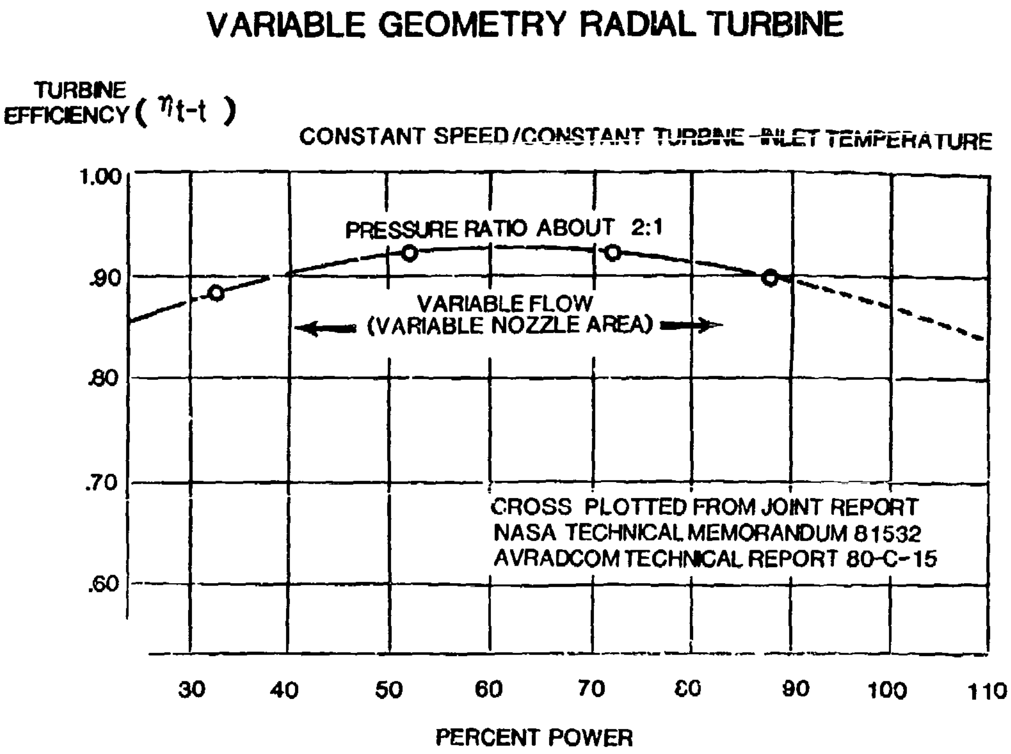

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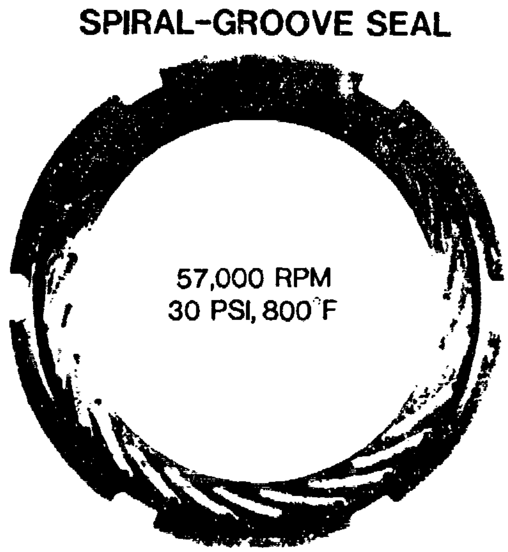

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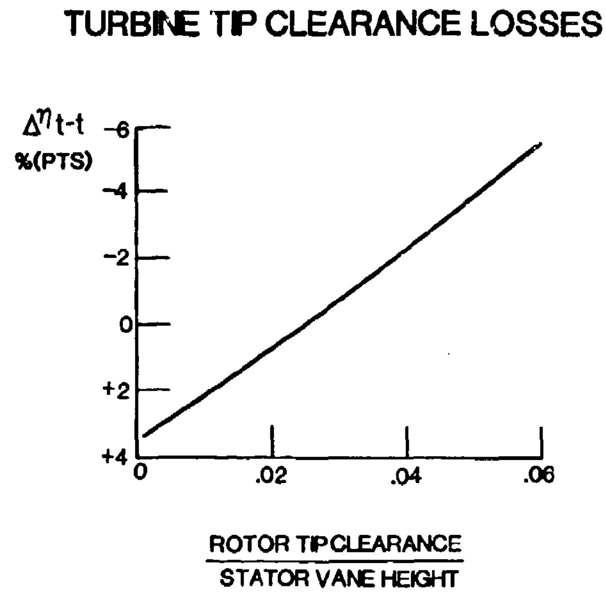

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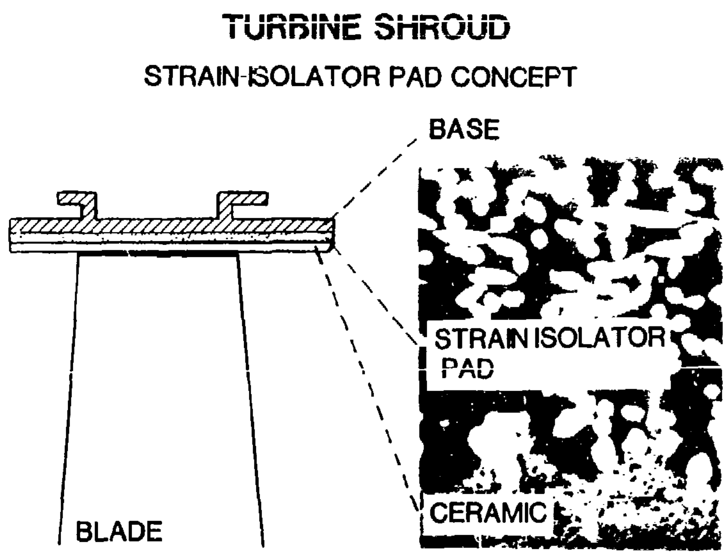

Figure 19

FUTURE REQUIREMENTS FOR HELICOPTER PROPULSION SYSTEMS

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SUMMARY

Requirements for propulsion systems have to be derived from the military missions. In this context the question of single or twin engine systems will be discussed.

The trend to more sophisticated engines is reviewed in the light of diminishing resources of materials and fuel.

Expected future economic conditions make it mandatory to counteract the increase of life cycle cost experienced in most of today's military systems. Means to reduce LCC are discussed.

INTRODUCTION

This paper intends to indicate which direction the development of future helicopter engines should take in the eyes of those responsible for gas turbine engines in the ministry of defence.

Future helicopters to be in service with the German Forces will have basic requirements for the propulsion system in common, because all of them, even the transport helicopters will operate for a certain portion of their flying time at the edge of the combat zone. For the following considerations the Anti-Tank-Helicopter is taken as a base line since it is placing the more challenging requirements on the propulsion system.

MISSION REQUIREMENTS

Envisaged area of operation and hence required flight profiles dictate the mission requirements for the helicopter engine. Due to their relatively low speeds and enemy use of advanced reconnaissance and detecting devices on the battlefield, helicopters will have to exploit any shred of cover the terrain offers. When flying a mission their height above ground will be determined by the type of cover available. A study of the major types of cover to be encountered in Germany yielded the distribution shown in figure 1. The various types of cover are between 10 and 40 m in height.

Taking into consideration that cover will be temporarily abandoned during a mission for search, aiming, and firing purposes the helicopter will operate at heights between 1 m and 50 m above ground. Its flight path will follow the terrain contours, remaining - wherever possible - 10 m below the available cover, since enemy radars can also see through thin tree tops.

A study on the antitank helicopter arrived at a total flying time of two hours and thirty minutes for the required mission; about one-third of this time would be flown under cruising conditions and two-thirds in a combat environment. Three speed ranges were determined: Range I covers hovering and flight at speeds of up to 50 km/h; range II encompasses flight at speeds between 50 and 150 km/h; and range III denotes cruising flight at 250 km/h including a portion of about 10 per cent at the maximum speed of 300 km/h. Figure 2 shows that 99 per cent of all flight situations occur at altitudes between 10 and 30 m, and 60 per cent at speeds of 150 km/h and below.

On account of these flight profiles future helicopters will operate during a large portion of their flying time close to the ground. This mode of operation makes it imperative to review existing data about engine failures caused by foreign object damage. The one year statistical table of engine failures in German Army helicopters (table 1) indicates that damage by foreign objects is a rather frequent occurrence. The table does not yet reflect the anticipated high ratio of future flights "nap of the earth" (NOE) or in the hovering mode. Therefore an increasing FOD rate has to be expected in the future. For this reason, provisions will have to be made that sand, stones, hail or ice, are separated from the air before it enters the compressor. An integrated Particle Separator as used by General Electric in their T 700 engine is considered an adequate solution.

The incorporation of foreign object separators may also be necessary for design reasons. If, for example, cost considerations lead to the installation of integral rotor stages in the compressor the risk of having to replace an entire rotor stage due to FOD on one blade is unacceptable. The use of compound materials such as carbon fibre in the compressor requires also protection against foreign object damage.

The scenario future helicopters will be operating in makes it mandatory that Infrared Radiation emitted from the hot parts of the engine has to be suppressed in order to reduce as much as possible the detectability of the helicopter by IR sensors und its vulnerability by IR homing missiles. Figure 3 shows how infrared signature is influenced by the temperature involved: Reduction of the exhaust pipe temperature from 640 K to 370 K reduces the IR signature to one per cent of its original value.

SINGLE OR TWIN ENGINE PROPULSION SYSTEM

This question - which for helicopters of the upper weight classes has been decided already in favour of twin or multiple engines - must be evaluated for medium and light helicopters on the basis of the mission requirements.

The arguments in favour of a single-engined helicopter are listed below:

- the propulsion system is cheaper and lighter,
- for that reason, the helicopter is lighter, smaller and cheaper,
- fuel consumption is lower, and
- the maintenance cost of one engine is lower.

It is conspicuous that this list does not include life-cycle costs. The reason is that they are dependent, among other things, on the loss rates which can only be estimated when the required mission profile and the area of operations are known.

According to what has been said above the following arguments in favour of the twin-engined military helicopter can be summarized:

- no "dead man's zone" (if an emergency rating is available),
- in the event of one engine failing:
  - safe landing,
  - self-evacuation capability,
  - possibly execution (or completion) of mission,
- lower loss rate - lower life-cycle costs (possibly),
- no need to optimize the rotor system for autorotation,
  - smaller rotor,
  - higher maximum flying speed (emergency rating),
- lower fuel consumption in stand-by mode, since one engine is shut down,
- smaller starting power required,
- in the event of FOD or exposure to enemy fire, higher probability of one engine remaining operational,
- easier handling, since engine weight is lower, modules are smaller,
- quicker accumulation of operating experience (engine running hours).

Figure 4 comes from an evaluation carried out with respect to the PAH-2/HAC antitank helicopter. It shows in terms of height above ground vs. I.A.S., the "dead man's zone" for both the single and twin-engined helicopter as a function of the ratio between the emergency rating of the remaining engine (after engine failure) and the required hovering power O.G.E. (outside ground effect).

The parameters are:

- pressure altitude: 1000 m,
- ambient temperature: 25 °C,
- time required to accelerate engine to emergency output: 1 second,
- ground impact speed: 6 m/s,
- pilot does not use the collective pitch after engine failure.

It can be seen that the advantages of the twin-engined helicopter can only be exploited if two requirements are met by the engine:

First, the engine must be designed for a very high emergency power. In this example, a calculated emergency rating of 140 % (relative to one half of the total needed O.G.E. hovering power) would be required for full elimination of the "dead man's zone". The emergency rating required may be less than this figure, since the kinetic energy stored in the rotor can be applied to reduce the impact speed. This is also a matter of optimizing the rotor mass.

The second important aspect is the acceleration capability of the engine. An extension of the response time from 1 second to 2 seconds would offset most of the benefits gained by the high emergency rating.

It is normal procedure for helicopter pilots not to fly within the dead man's zone longer than absolutely necessary. However, under simulated combat conditions an - unacceptably - large portion of the mission would have to be flown within the dead man's zone. From this it is obvious that flight training in peace-time could not be performed with a single-engined antitank helicopter if flight safety aspects shall be observed.

CHANGED PARAMETERS

It is becoming obvious that, in the long term, some of the parameters applicable to the development and operation of our defence equipment will change for the worse. Two aspects shall be covered in this context: fuel and material.

In the past, the design engineer could well assume that aviation fuel of high quality would be available for the operation of the engine he was to design. But the situation is changing. In 1980, the freezing point of civilian jet A-1 fuel was raised by three degrees to -47 deg. Centigrade. Although this change will so far not have any great direct influence on military flying operations (its impact on long-distance flights and in-flight refuelling is still being examined) we must adapt in time to the fact, that as a long-term trend, the quality of aviation fuel will deteriorate.

Furthermore, future engines will have to be able to burn alternative types of fuel. It is known, however, that synthetic hydrocarbons extracted from coal, oil shale or tar sand have, by comparison, considerably poorer properties than our present products. Their higher contents of aromatics produce a higher radiation intensity of the flame and this, in turn results in a higher load on the combustor walls. In future, combustor exit temperatures might be further increased, if possible, to improve the thermodynamic cycle. This should be done with alternative fuels in mind, the properties of which, however, are not exactly known yet.

The hope on metallurgical advances bringing about better and better materials for combustors may not be realistic, since some important alloying elements are even now getting scarce (e.g. cobalt in combustor material C 263). Therefore, future engine design will have to rely more on technology than on materials.

Since weapon systems are kept in service increasingly longer, engines which start development now will have to operate still in the next century when optimum types of fuel or material may not be available. This means e.g. for the hot section of an engine that cooling techniques will have to be improved.

Figure 5 shows a rather old but interesting example of turbine blade cooling: The Heinkel-Hirth 109-011 turbojet engine of 1942. The designer, Dr. v. Chain, emphasized that no nickel or other high-temperature materials were used anywhere in the engine, as such materials were no longer available in Germany at the time. Thus about 30 % of the compressor discharge air was used for cooling.

Future fuels with poorer properties will similarly call for improved cooling technology. Work on methods to reduce the cooling air temperature will probably be a step in the right direction. Thermo-coating might also enable a cheaper material to be used for combustors.

Moreover, we must pursue the work on the use of ceramics in the hot section of the engine, even though a breakthrough in this field may not be possible in the near future. Work should be directed towards static hot parts, where new concepts could create problems for metals e.g. in an engine with recuperator, where combustor wall cooling has to be achieved at cooling air temperature levels far above compressor discharge temperature.

Another very important material for aircraft engines is titanium, a material which should perhaps also be regarded as being in short supply. Figure 6 shows the upward trend of the titanium price, using a component of the LARZAC engine as an example. It is difficult to say whether the price trend, seen by itself, is a sufficiently conclusive indicator of the scarcity of the material, but the price alone should be an adequate incentive to intensify work on high tensile strength steels for compressor rotor discs.

In addition, manufacturing techniques for titanium should be reviewed. The final titanium part in an engine after machining makes up only about 20 % of the raw material used, while 80 % are non-recyclable scrap. Since powder metallurgy for superalloys is widely used in aircraft engines, the application of this technique also to titanium should be intensified. The price of titanium parts can be expected to be reduced by up to 80 %, not to mention the large energy savings by use of the powder metallurgy process.

LIFE-CYCLE COSTS

A comparison of military technology of the East and the West indicates that the Soviet Union has so far invariably stressed numerical superiority and has, therefore, developed its weapon systems in the shape of simple, rugged items and procured them in large numbers. In spite of this philosophy it has achieved very good technological results in many fields, including that of aero-engine construction.

The West has logically not attempted to compete in the field of large production numbers, but has endeavoured to apply the most advanced technology to its weapon systems.

This approach has provided high performance to our airborne weapon systems, but also made them sophisticated and very costly.

This trend coincides unfavourably with the negative development of most economies of the Western nations. This development is largely attributable to the explosion of the oil prices and makes it impossible for the time being to absorb substantial cost increases in the defence budget, for example, by high growth rates of our gross national product.

It follows that we have to make every effort to come to grips with the cost problem of new weapon systems and to examine, at this point, what share the engine may contribute to this endeavour.

Figure 7 shows a breakdown of the life-cycle-costs of helicopter engines. Engine definition, though its share of costs is not made visible separately in this figure, has the greatest impact on life-cycle costs because about 80 % of total LCC are committed at this early stage of a program.

Engine concept and thermodynamic cycle have to be selected in view of the helicopter's mission and the power requirements. Careful trade-offs have to be made between high specific performance, sophisticated design, lowest fuel consumption on the one hand and simplicity of design, lower production and spare parts' prices, easier maintenance on the other.

Taking again an engine in the 750 kW class as, for example, for the antitank helicopter, parametric studies showed that optimum turbine entry temperature is in the 1400 K range and pressure ratio about 12. Any increase in specific performance would lead only to a decrease in airflow with all problems of tolerances and clearances in a small engine and related loss of efficiency.

In this context the provocative question may be allowed, to what degree narrow design tolerances for seals, tip clearances etc. are worth the related production cost increase, since in most cases a part of the performance gain is lost to degradation after only a few running hours.

The aforementioned values for pressure ratio (12) and turbine entry temperature (approx. 1400 K) make it a realistic goal to design future helicopter engines of this power class having very few rotating parts and possibly uncooled turbine blades. Figure 8 shows that industry is working already in this direction.

Two other aspects should be mentioned: The number of modules and the design of engine families.

The modular design yields a number of advantages and cost reductions especially in the spare engine and maintenance area. However, the number of modules should be limited because every additional interface causes a weight and cost increase. Five to six modules should be enough for this type of engine.

In view of the high development and production cost in relation to the relatively small number of military engines produced the principle of designing an engine family for military and civil application as well appears advisable to reduce the unit production price.

The users have become very sensitive to fuel consumption. To clarify the order of magnitude the fuel has within LCC for a military helicopter a comparision with the civil helicopter is helpful. The main difference is in the usage rate of 3000 hrs/year for the civil and 250 hrs/year for the military helicopter.

Under the assumption that a 10 % reduction in SFC to be designed into an engine would increase the production price by 10 % only, at today's fuel prices the civil helicopter would break even after 2 years. The military helicopter accumulating only 1/12 of the civil flying hours per year would need 24 years, i.e. it could not break even in the 20 year service life of the system. It is evident, that fuel consumption of military helicopter engines does not have the same priority as for turbojets. This does not mean, however, that the goal of low SFC could be neglected. All measures to achieve low SFC should still be pursued because with fuel prices heading out of sight (figure 9) it will be difficult in the future to keep flying hours at a level necessary to maintain proficiency. Means to reduce SFC shall, therefore, be discussed.

Figure 10 shows the power requirements of a helicopter vs. I.A.S. It can be seen, that on average total mission fuel will be burned at a power demand of about 60 % of max. take-off power. Therefore, fuel consumption should be reduced particularly in the partial power range. This could be achieved by different approaches.

Component efficiency

A lot of research work is going on to increase efficiencies of the engine's turbo machinery components because of their great influence on overall performance. However, from the high level of technology already obtained it will be difficult to achieve large improvements in this field. Cooled high-pressure turbines may have some efficiency potential provided we succeed in preventing, at least in part, the loss of turbine efficiency

caused by the mixing of the cooling air into the gas stream. Other measures such as the decrease in the quantity of cooling air by precooling or by limiting the amount of cooling air during partial load operation may not be reasonably applied to this size engine.

Variable geometry components

In a conventional gas turbine engine the main reason for the increase of specific fuel consumption at partial load is that it operates at a lower pressure ratio compared to full power. The goal of variable geometry turbo components is, therefore, to keep pressure ratio and turbine entry temperature at design point levels even at the reduced part-load airflows. Variable geometry in the compressor and the power turbine are state of the art today, however, these alone do not yield significant improvements. Flow variation at the same time by means of a variable gas generator turbine appears difficult because of its high temperature level.

In 1978, the Aix-La-Chapelle Technical University conducted a study on behalf of the Federal Ministry of Defence to determine what reduction in the fuel consumption for partial load operation could be accomplished by the use of a two-stage gas generator turbine with a variable second stage. Using an antitank helicopter mission as a basis, the study arrived at fuel savings of 5.5 %. In view of this small saving the expense of designing variable geometry into all three turbocomponents might not be justified.

Gas turbine engine with recuperator

There are two ways, in principle, of integrating a recuperator into a turboshaft engine. In figure 11 the recuperator is installed in the usual way downstream the power turbine. Figure 12 shows the specific fuel consumption of this engine versus the pressure ratio, the turbine entry temperature being the parameter. The diagram is calculated with constant heat-exchange ratio of the recuperator and constant design power level of 900 kw. In contrast to the conventional gas turbine engine without recuperator turbine entry temperature has a significant influence on specific fuel consumption. In addition minimum SFC's occur at considerably lower pressure ratios.

It can be seen that the turbine entry temperature should be kept on a high level even at part power ratings. Pressure ratio may decrease a little bit, assuming that the exchange rate of the recuperator increases at the same time. Variation of the thermodynamic cycle in the engine is achieved by variable guide vanes in the power turbine and the related variable geometry in the compressor.

Figure 13 shows a different arrangement of the recuperator. This time it is located between the gas generator turbine and the power turbine and is bypassed during full- and emergency power through a variable bypass. Since this arrangement has not been studied in detail yet, it cannot be fully assessed, whether it will be possible to realize the expected advantages.

Advantages over the first arrangement could be:

- large amount of heat exchangeable even at moderate turbine entry temperatures: simpler turbine cooling,

  less volume recuperator because of higher pressure level of compressor discharge air (higher optimum pressure ratio) and gas stream as well,
- gas generator operation in the upper half of part power range at constant speed and near optimum values for pressure ratio and turbine entry temperature which remain almost constant: less cycle fatigue, rapid acceleration,
- emergency power without limitation by recuperator,
- no variable geometry necessary in turbo components.

Disadvantages could be:

- variable bypass required: expensive and design critical bypass valve in the hot section between turbines,
- SFC reduction at part load only,
- IR-Suppressor same size and weight as for conventional engine.

Only few findings are available as yet on helicopter engines with recuperators. In 1978/1979, however, detailed studies on tank gas turbines with recuperators were conducted by KHD Oberursel and MTU Munich on behalf of the Federal Ministry of Defence.

We will have to discover what findings from the work on the tank gas turbine are also valid for helicopter engines. For the near future we shall be unable, for budgetary reasons, to develop a completely novel helicopter engine with a recuperator.

The regenerative engine will probably not be competitive on just lower fuel consumption. However, it could have a chance if it leads to a lower weight of the propulsion system and thus, considering the weight factor, to a lighter helicopter.

REDUCTION OF ACQUISITION AND MAINTENANCE COSTS

To reduce production costs, to which spare parts' prices are closely related, the following requirements have to be implemented, some of which have been incorporated in the specifications for the Antitank Helicopter 2:

- minimize number of parts and modules,
- minimize material consumption,
- use wherever possible materials that are low in price while making due allowance for life-cycle requirements and costs,
- minimize amount of machining,
- aim for recycling (as earlier discussed for titanium),
- minimize amount of handling, and
- minimize number of inspections during production.

Today still too much machining is done which makes for high costs and causes a waste of raw materials. Machining should be replaced by other and cheaper processes such as casting, that will also economise on materials. A case in point is a diffusion-bonded turbine rotor consisting of a hot-isostatic-pressed MV 1460 disc and a cast IN-100 blade assembly (figure 14).

Finally, efforts should be made to reduce the number of inspections during production which are very time-consuming. The manufacture of components in one step, for instance, would call for only one final inspection as compared with one or more intermediate inspections required during the machining process.

MAINTENANCE

One of the principal requirements to ensure cost-effective operation of weapon systems is their ease of maintenance. This requirement caused the modular design technique and the "on-condition" maintenance, whereby engine components are replaced according to the real life time consumed.

Assemblies must be easily accessible and simple to separate. (For example the change of the flame tube without disassembly of the turbine). Maximum use should be made of quick-disconnect couplings (clamping bands) instead of bolted joints, a method that also could be used to fasten accessory as shown in figure 15. Efforts should be made to simplify the replacement of modules or assemblies. It would be desirable if a description on the equipment itself would suffice to replace it, thus reducing or not requiring at all the use of manuals or specially trained technicians.

The above mentioned easy replacement of modules and accessories by design should lead in turn to:

- minimum number of standard tools required for infield service,
- less voluminous handbooks,
- less training for maintenance personnel,
- less maintenance manhours per flight hour (MMH/FH).

As a last point the use of a digital control unit shall be emphasised. Besides a better control of the engine by more and quicker registration and conversion of the environmental parameters to corresponding engine setting and hence more efficient use of the engine's potential, the digital control unit could also be used as part of a maintenance recorder, a necessary device for on-condition maintenance. As the control unit registers already some engine parameters as RPM, TET, etc. the free computer capacity could be used to calculate the component life actually expended and give a warning in case of failure or before design life is fully consumed.

The use of a digital fuel control unit (DCU) makes it advisable to separate it from the engine, as engine bay temperatures of about 250°C and high vibration levels may cause a malfunction of the electronics.

The future DCU will not be a part of the engine anymore. Engine- and module changes should be possible then without changing the control unit. In this case self-adjusting fuel control units are needed to avoid time-consuming and costly test runs for adjustment. Use of DCU places a new requirement on the engine/fuel control designer. That is to harden the electronics against the electromagnetic pulse of nuclear weapons (EMP). All advantages of the electronic fuel control would be reversed to the contrary if the vulnerability of the system would be increased.

CONCLUSIONS

It appears logical, that in view of changing economic parameters worldwide, future helicopter engines should be developed along the lines of

- non sophisticated concepts,
- cautious cycle selection in view of small engine size,
- development risk to be reduced by technology programs,
- simplicity of design, with fewer parts and modules,
- most economic manufacturing processes to achieve low production costs, low spare parts' prices,
- high reliability i.e. more early testing when discovery of problems is less costly,
- good maintainability.

Advanced technology should be applied on the next generation engines to improve LCC instead of achieving pure performance. Thus not the performance related technological limits or properties appear to require improvement in the first place but their cost driving factors which will decide whether we can afford military systems in the future. The question will inevitably arise of how the certainly larger front end investment on better engine design, development and more thoroug testing to get all those properties into an engine shall be paid for. For the time being, the answer should be: more international cooperation.

ACKNOWLEDGEMENTS

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ENGINE-RELATED INCIDENTS	
ENGINE FLIGHT HOURS	190.400
NUMBER OF INCIDENTS	97
UNEXP. ENGINE REMOVALS	32
FOREIGN OBJECT DAMAGES	24
ENG. COMPONENT FAILURES	15
ACCESSORY FAILURES	10
OTHER EVENTS (CHIP WARNING, etc)	50

Table 1 Evaluation of Engine-Related Incidents

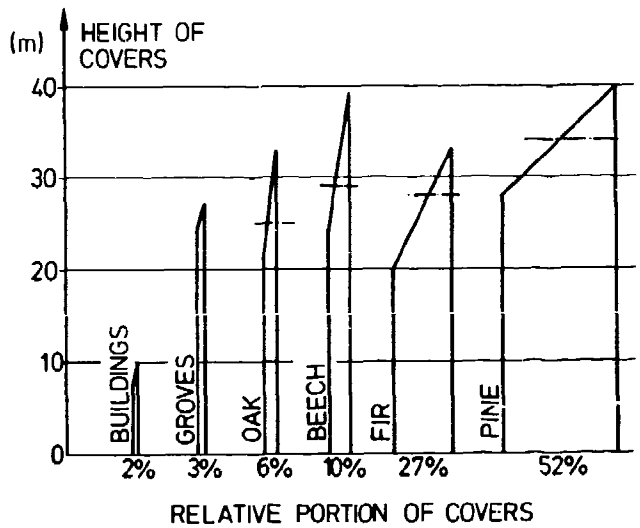


Fig. 1 Types of Covers

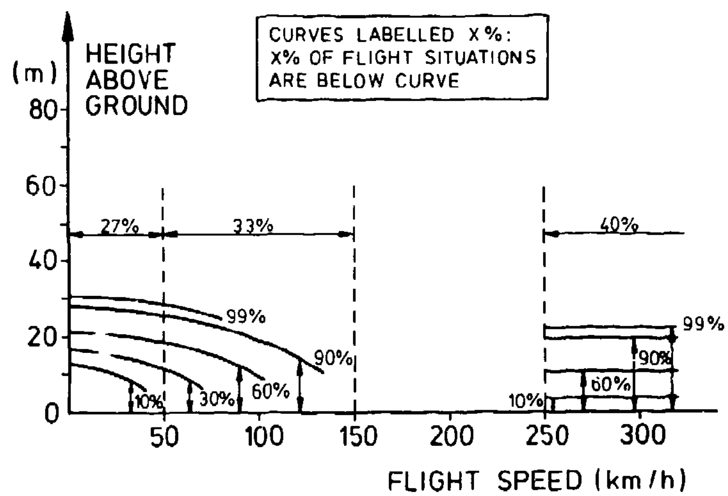


Fig. 2 Distribution of Flight Situations

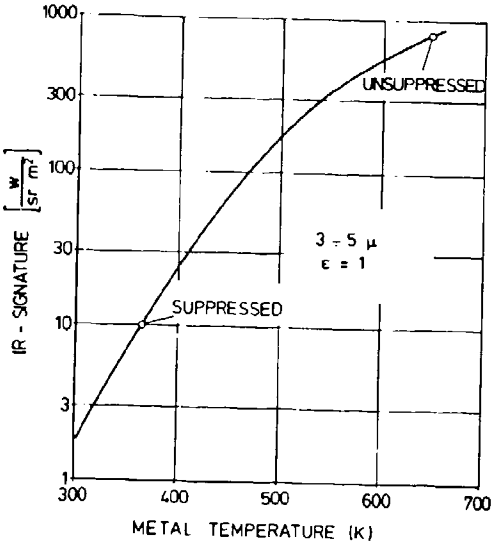


Fig. 3 Temperature Influence on IR-Signature [3]

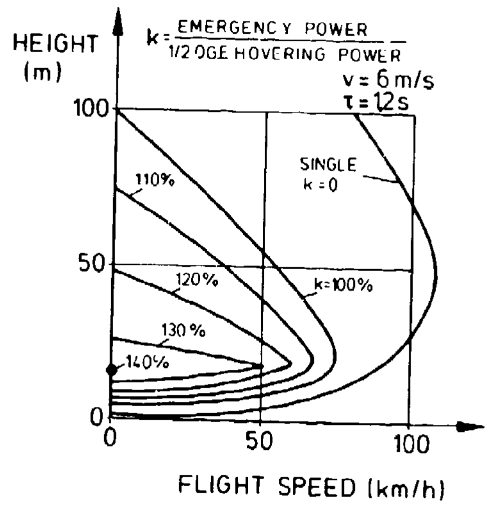


Fig. 4 Dead Man's Zone

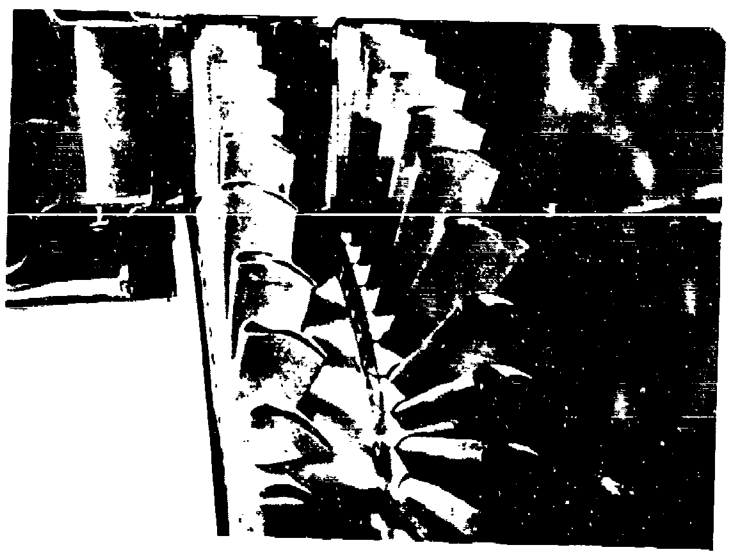


Fig. 5 Heinkel-Hirth 109-011 Turbojet Engine (Courtesy J.Bush, Aero Prop. Lab. USAF)

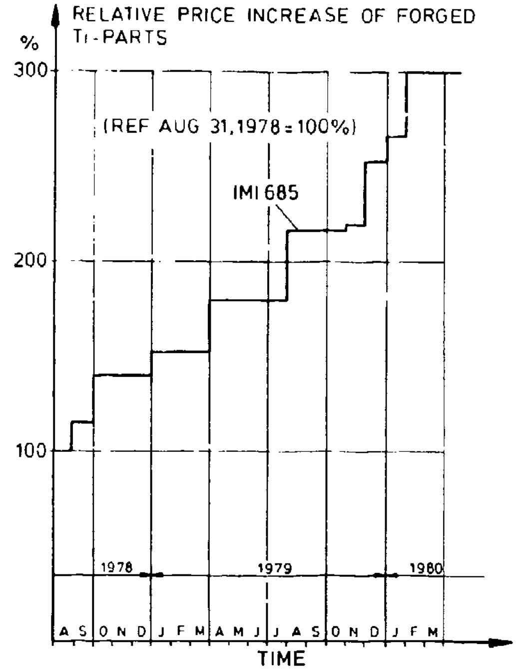


Fig. 6 Titanium Price Increase

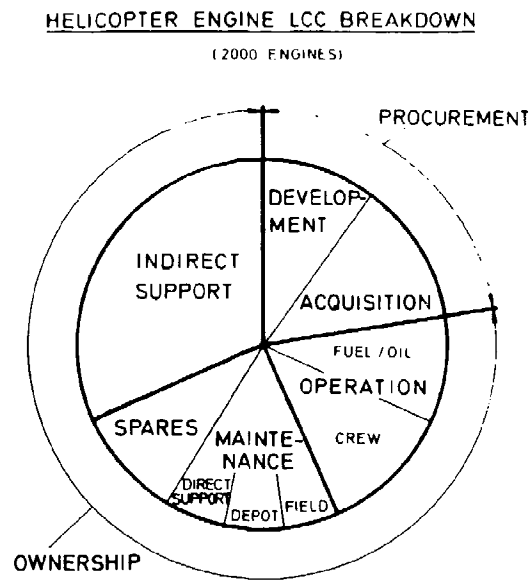


Fig. 7 LCC of Helicopter Engines [1]

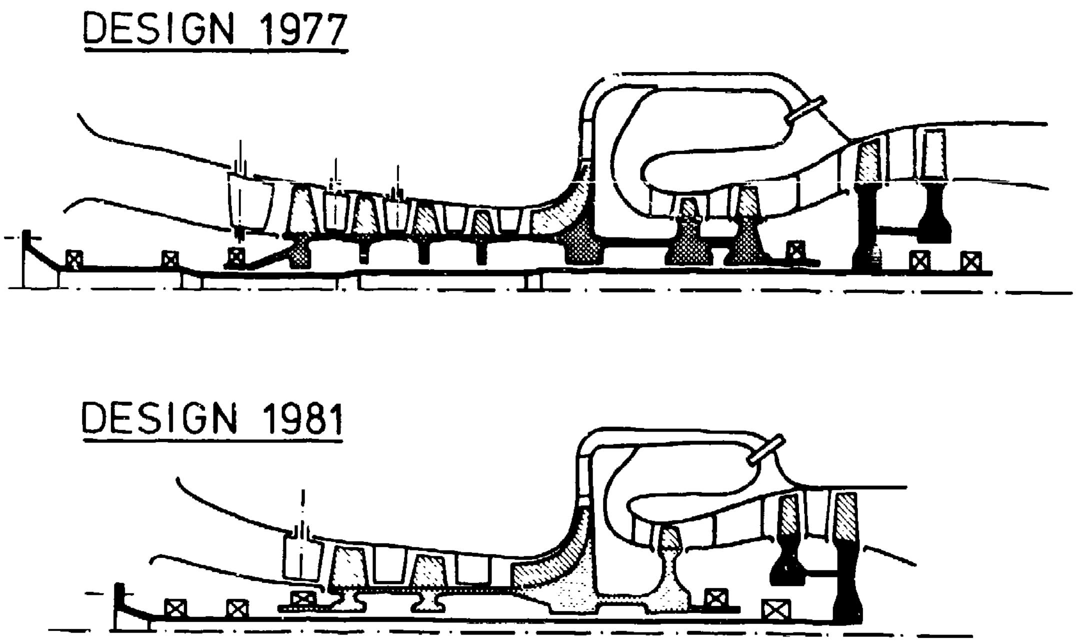


Fig. 8 Engine Concept Improvement

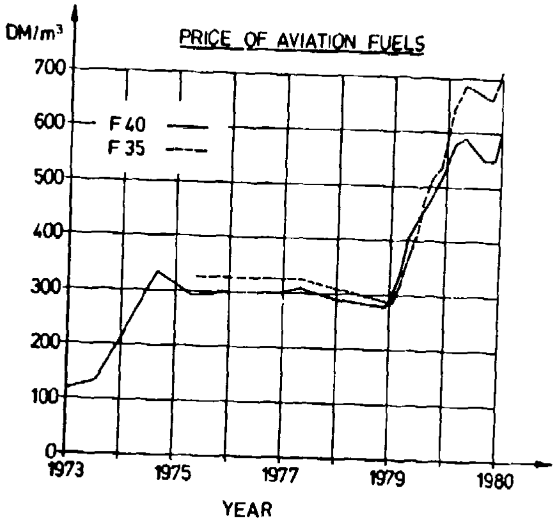


Fig. 9 Fuel Price Increase

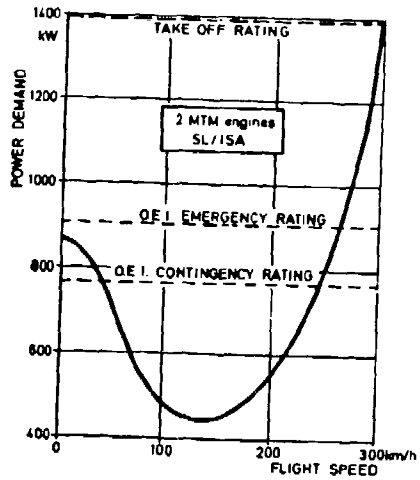
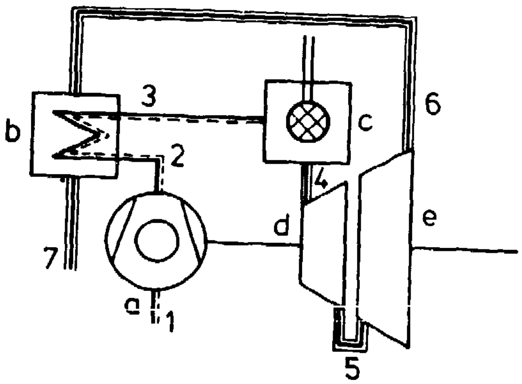


Fig. 10 Power Demand of the PAH 2 / HAC



a COMPRESSOR
b RECUPERATOR
c COMBUSTOR
d GASGEN.-TURBINE
e POWER TURBINE

Fig. 11 Recuperator Downstream the Power Turbine

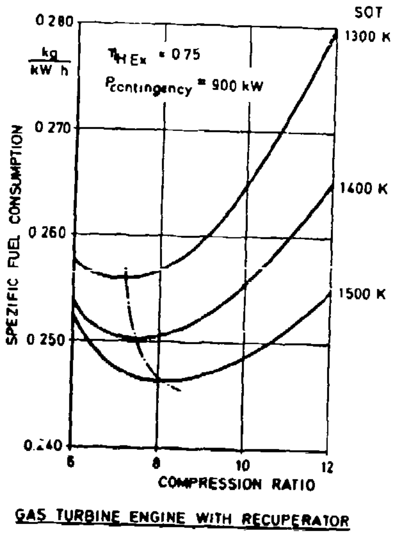
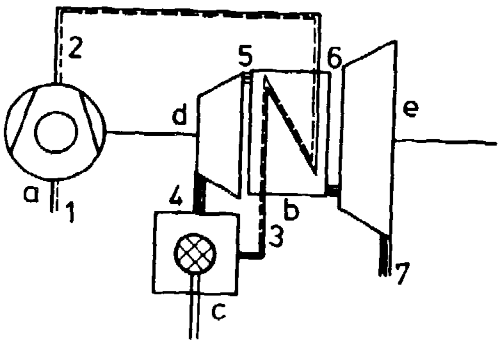


Fig. 12 SFC of a Gas Turbine Engine with Recuperator



a COMPRESSOR
b RECUPERATOR+BYPASS
c COMBUSTOR
d GASGEN TURBINE
e POWER TURBINE

Fig. 13 Recuperator Upstream the Power Turbine

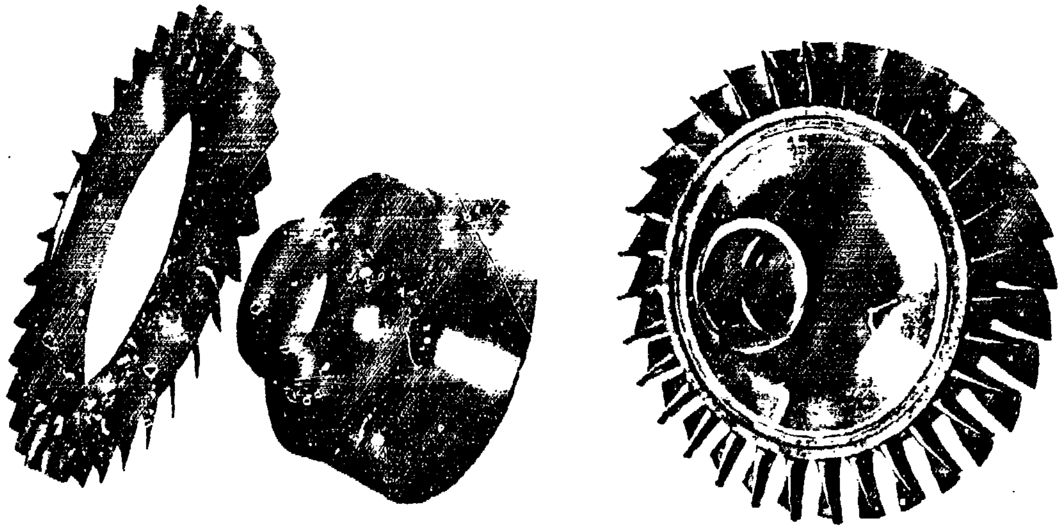


Fig. 14 Diffusion-Bonded Turbine Wheel (MTU)

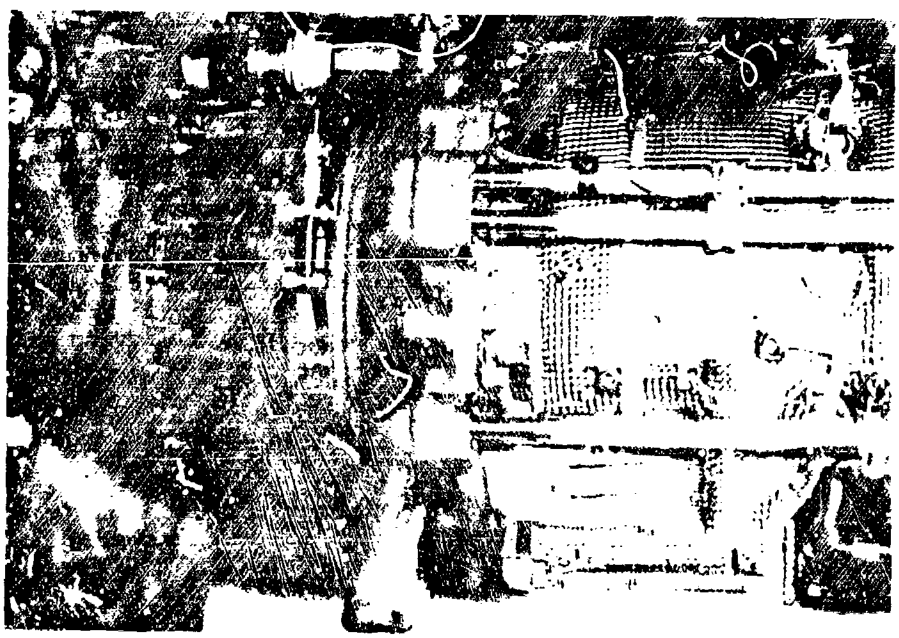


Fig. 15 Clamping Band Fitting of the T 112

DISCUSSION

H.Saravanmutto, Ca

Could you comment on the recuperated configuration where the heat exchanger is located between the gas generator and power turbine? Concern is that temperature level at this point would be much higher than that previously used.

Author's Reply

This configuration has been examined from first considerations only. A study is being initiated that will further define its potential.

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14. Abstract

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The Specialists' Meeting was organized into 6 sessions: Helicopter Propulsion Surveys (3 papers); Engine Component Technology (5 papers); Drive Train Component Technology (4 papers); Inlets and Particle Separators (2 papers); Engine-Airframe Dynamic Compatibility (4 papers); Future Advances and New Requirements (3 papers), and a keynote paper was delivered at the beginning of the meeting.

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